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PRESSURIZATION SYSTEM
FOR USE IN THE
APOLLO SERVICE PROPULSION SYSTEM
CONTRACT NAS 9-3521
PHASE I INTERIM REPORT

July 1965

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FOREWORD

This document is submitted in accordance with Article XI, sub-paragraph C, of Contract NAS 9-3521, dated 14 October 1964, received 21 October 1965.

CONTENTS

I. INTRODUCTION

This report summarizes work performed during the Phase I portion of Contract NAS 9-3521, "Pressurization System for Use in the Apollo Service Propulsion System." This contract is under the direction of the NASA-Manned Spacecraft Center in Houston, Texas.

Purpose of the contract is to develop an advanced, lightweight pressurization system (AIPS) for use in the Apollo Service Propulsion System. The requirements which must be fulfilled by the AIPS are

1) it must be compatible with the current Apollo Service Propulsion System,
2) it must offer a substantial weight savings over the pressurization system currently in use, and
3) it must be at least as reliable as the pressurization system currently in use.

The contract is categorized in two Phases. Phase I required design, analytical, and experimental efforts devoted to various candidate system concepts; and culminated in the selection of an optimum advanced pressurization system for the Apollo SPS application. Phase II requires the fabrication, assembly, and testing of a prototype of the optimum design selected.

The Phase I effort progressed in the following manner. The initial effort was a survey of existing pressurization systems and system concepts. From this survey, seven candidate concepts were selected for preliminary study. The preliminary studies concluded with the selection of three of the candidate systems to be subjected to more detailed design, analysis, and investigation. The detailed study effort included both analytical and experimental work, to assist in system design and the investigation of problem areas. Phase I then culminated with the selection of a single system to be built and tested as a prototype in Phase II.

II. SURVEY OF PRESSURIZATION CONCEPTS AND RELATED SYSTEMS

The Phase I work effort was initiated with a survey of propellant tank pressurization concepts and systems which could be applicable to future use with the current Apollo Service Propulsion System. The investigation included concepts involving technical considerations beyond present state-of-the-art; the major criteria for inclusion of systems in the survey were potentiality of weight savings, and compatibility with Apollo Service Propulsion System mission and vehicle requirements.

The survey produced nine basic pressurization systems for consideration. Several of the techniques involved had been successfully applied to operational propulsion systems or had progressed to developmental status. Others were purely conceptual in nature, with no history of detail design or testing at the time the survey was conducted.

Each of the nine systems is summarized and illustrated schematically below. Since actual hardware considerations are not pertinent to this discussion, and since system-to-system component requirements are very similar, small components such as valves, regulators, switches, filters, etc., are not shown in the system schematics. Although the systems discussed represent the basic concepts studied, variations in these systems were also included for consideration. For instance, fluids other than helium were also considered as pressurants, and heat sources other than a bipropellant gas generator were investigated.

Each system presented has been assigned a number for the purpose of convenience in referencing. The present Apollo pressurization system, being the reference system to which all others will be compared has been assigned the number "O" (zero).

A. SYSTEM O - AMBIENT STORED HELIUM
(Present Apollo Service Pressurization System)

The system shown in Figure 1 represents the present Apollo Service Propulsion Pressurization System. This basic system is also used in other present-day propulsion systems in both launch vehicles and spacecraft. Helium stored at high pressure and ambient temperatures expands through propellant feed line heat exchangers before entering the propellant tanks. Purpose of the heat exchangers is to nullify the cooling effect of helium expansion from the storage container. This system is basically the type used in the Agena, Titan III Transtage, and Ultra-Low-Pressure Rocket vehicles. The ambient stored helium system is a highly reliable pressurization system, due to 1) system simplicity, and 2) absence of extreme (high or low) operating temperatures. This also is a relatively inexpensive system to design, develop, fabricate, and maintain, again as a result of simplicity in concept and environment. The significant disadvantages of this system are its size and weight. Ambient storage of helium requires a considerable volume in relation to the overall system envelope. The containment of a large gas volume at high pressure results in extremely heavy storage containers.

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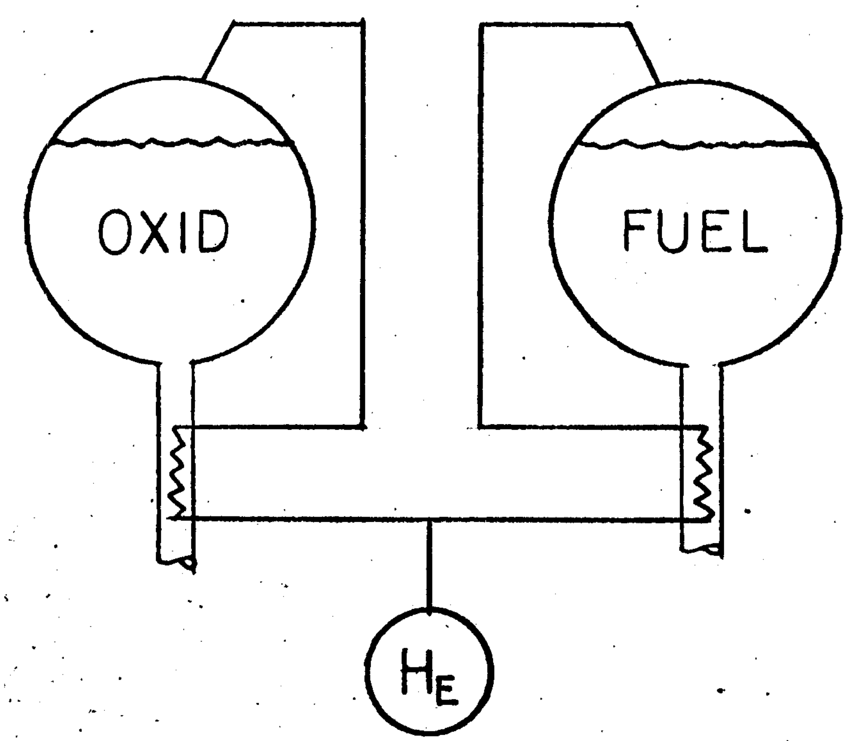
FIGURE 1 APOLLO SERVICE MODULE



B. SYSTEM 1 - CRYOGENIC STORED HELIUM

This system, shown in Figure 2, is observed to be similar to the ambient Stored Helium System, the only difference being in helium storage environment. In a vehicle where cryogenic storage is possible, a considerable reduction in weight can be realized by storing the pressurant cryogenically, then heating it prior to entry into the propellant tanks. The system shown uses the main propellants to heat helium to near ambient temperature. Thus, the helium storage density is high, affording a relatively small container, while the tank entering density is low. Another advantage of storing the high pressure gas at lower than ambient temperature is the strength-temperature relationships of the usable aluminum and titanium alloys which show a very significant increase in strength with decreasing temperature. Storage container weight can thus be reduced on two counts - container volume reduction and container material strength increase. The increased material strength at lower temperatures was used in the preliminary studies described in Section III. However, in the detailed design and analysis (Section IV) all pressurant storage containers were stressed for ambient temperature (530°R) allowables.

C. SYSTEM 2 - CRYOGENIC STORED - HEATED RESIDUAL HELIUM

A method of reducing helium storage volume, and therefore storage weight, of the system previously discussed, is to heat the storage container residual gas as shown in Figure 3. The optimum controlled heating profile yielding minimum system weight can be

FIGURE 2 STORED HELIUM

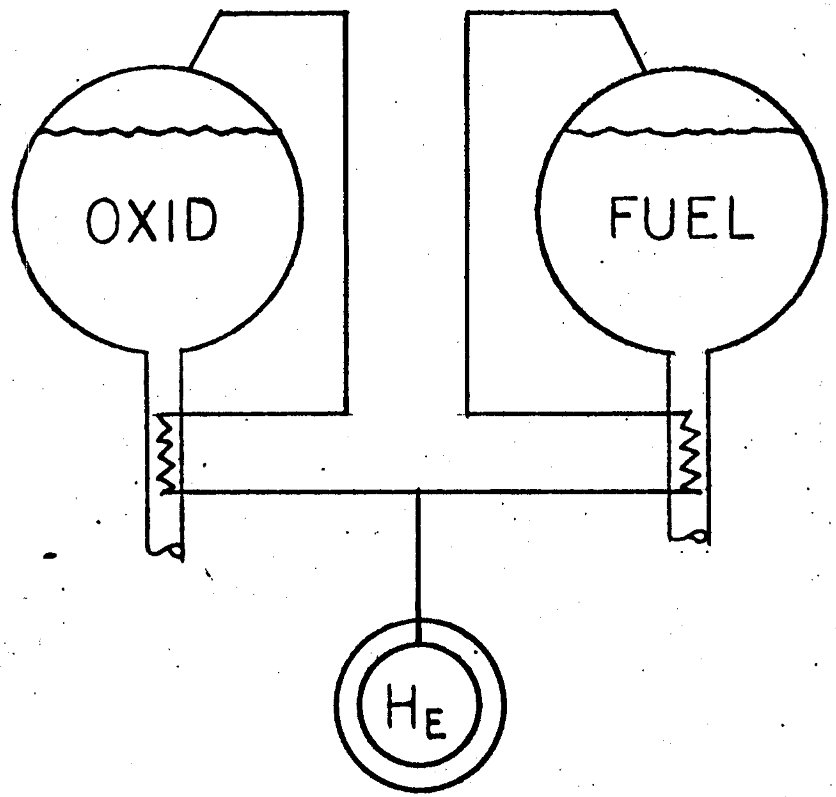
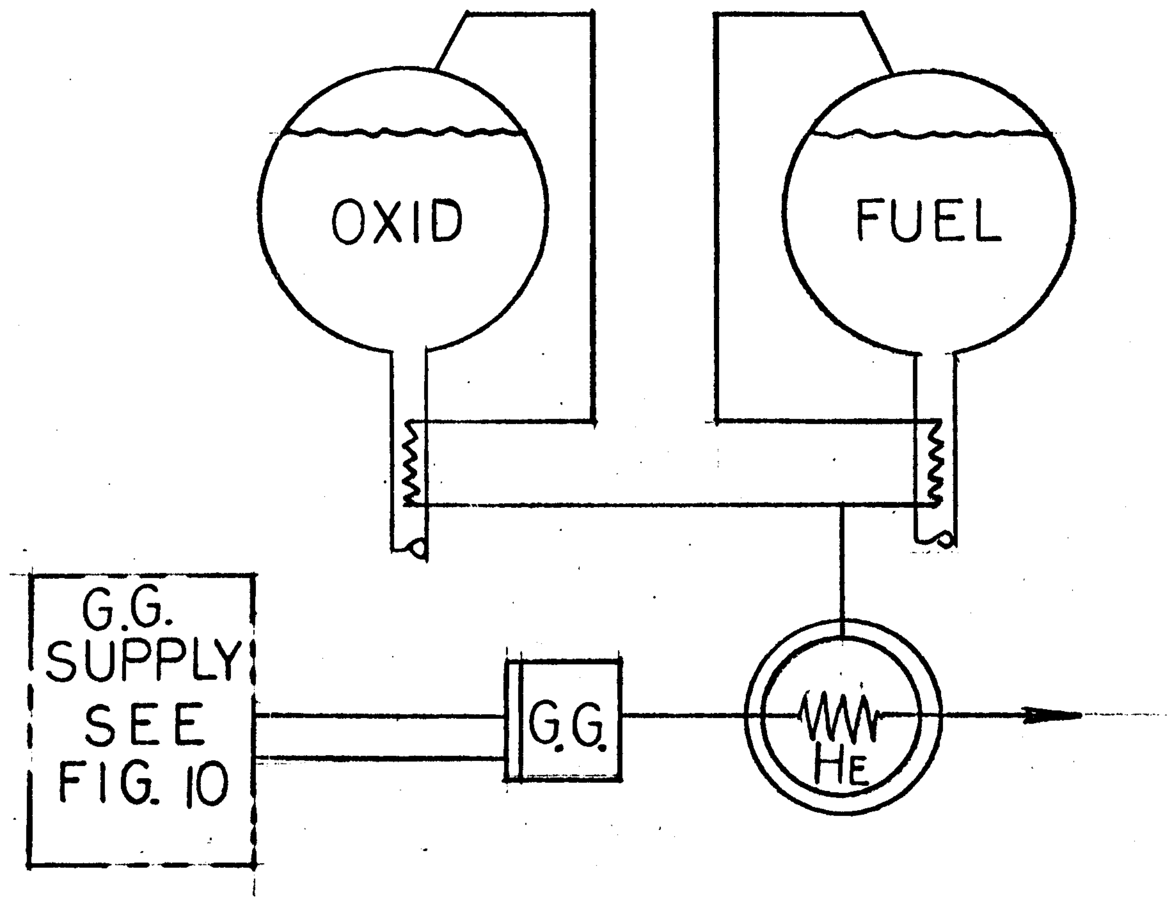


FIGURE 3 STORED HELIUM - RESIDUAL HEATING

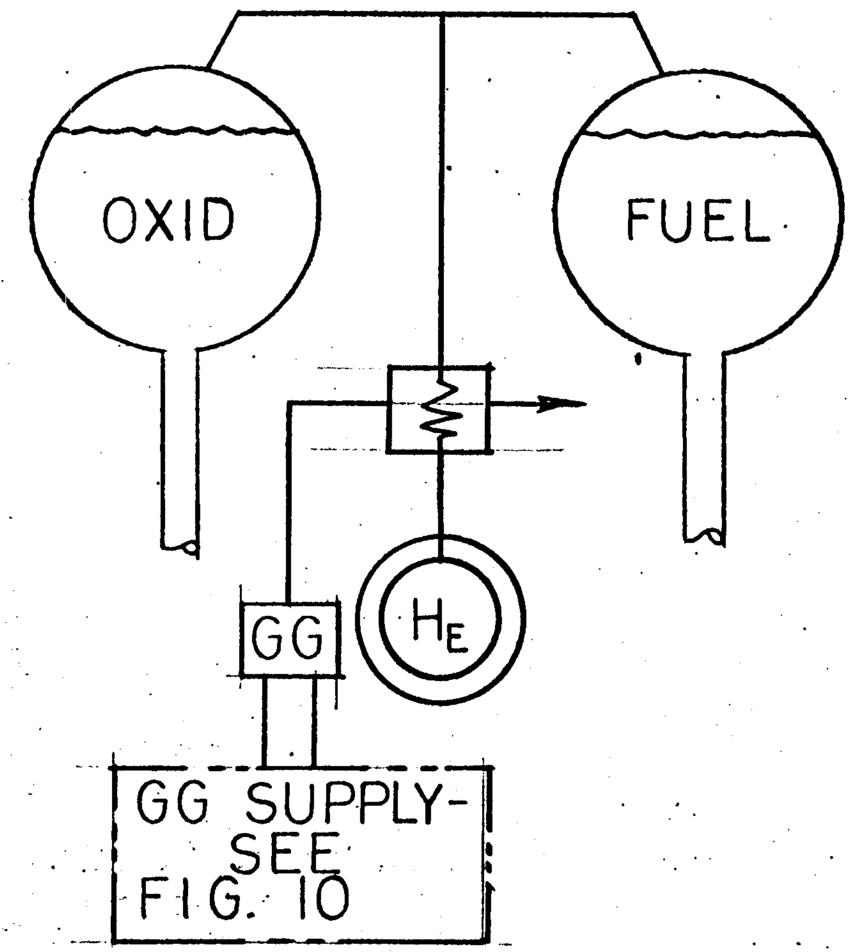


analytically determined. The philosophy illustrated by this technique is that although <ins>initial</ins> helium storage conditions of low temperature and high pressure are conducive to minimum system weight, a <ins>final</ins> gas state of high temperature and low pressure within the storage container has the effect of reducing the mass of unusable residual helium and therefore the mass of the initial helium load. As shown in the figure, a gas generator may be incorporated as a heat source. Since heating of the storage container in this way may be comparatively low level, it is anticipated that auxiliary propellant feed line heat exchangers should be used to increase helium temperature to ambient.

D. SYSTEM 3 - CRYOGENIC STORED - HEATED HELIUM

The system in Figure 4 employs helium, stored cryogenically, passed through an active heating system enroute to the propellant tanks. Purpose of this technique is to heat the entering pressurant to a relatively high temperature, which reduces pressurant density and thereby minimizes the total mass of helium required for propellant tank pressurization. Elevated helium temperatures can be attained more efficiently by this method than by heating the storage container. Also, it may be advantageous to maintain a higher-than-ambient ullage temperature during engine burn periods so that the cooling effect during coast will tend to decrease the tank pressure. This can greatly reduce, or eliminate, tank pressure overshoot due to coast period heating, and decrease the weight penalties associated with venting and/or increased tank design pressure. To be most

FIGURE 4 STORED HELIUM-ACTIVE FLOW HEATING



effective for multi-coast missions, this system must be capable of providing hot gas to the propellant tanks at all times, including prior to main engine start when tank pressures are below operating limits.

E. SYSTEM 4 - CRYOGENIC STORED - HEATED RESIDUAL - HEATED HELIUM

This system (Figure 5) combines the use of residual pressurant heating (system 2) with active helium heating in the supply line (system 3). It combines the advantages of low pressurant mass requirement and minimum pressurant storage residual. The gas generators, used for heating, could also be combined into a single unit, depending upon the relative hot gas flow requirements to each heat exchanger and the resulting control problems encountered.

F. SYSTEM 5 - CASCADE HELIUM STORAGE PROPELLANT FEED LINE HEATING

The cascade storage system (Figure 6), like system 2, was conceived in an effort to reduce helium residual mass and the associated primary storage container size. This is done by replacing the cold helium flowing out of the primary container with warmer, less dense helium from an ambient temperature secondary container. The warm and cold gases in the primary container are separated by a flexible membrane (the membrane was later deleted, as discussed in Section V). The main pressurant is expanded through propellant feed line heat exchangers to bring the temperature up to near ambient. The weight saved in the primary storage container must be compared to the weight added by the secondary container to determine desirability of the cascade concept.

FIGURE 5 STORED HELIUM-RESIDUAL/ACTIVE HEATING

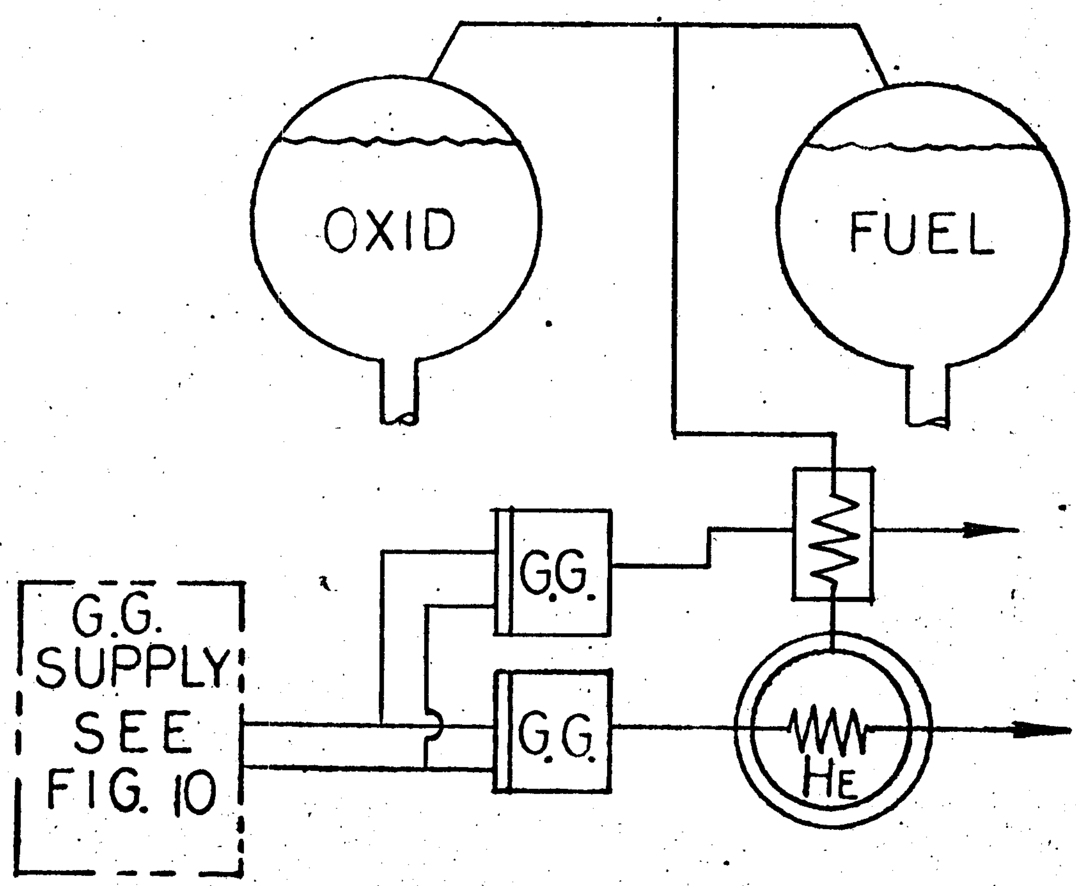
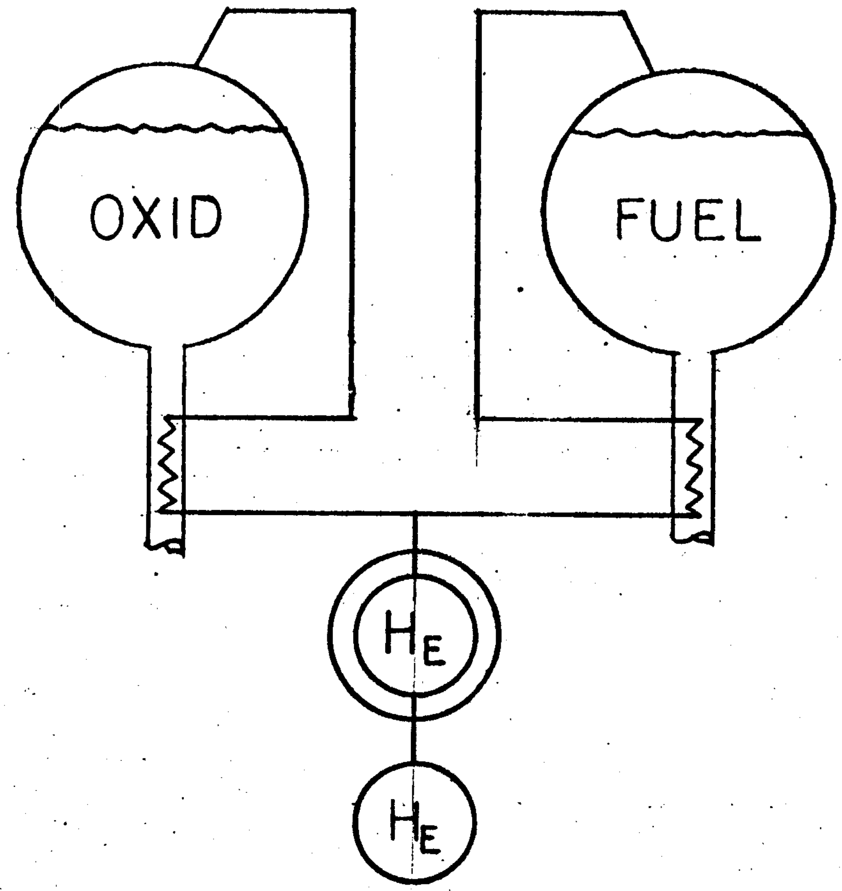


FIGURE 6 CASCADE-PASSIVE FLOW HEATING



G. SYSTEM 6 - CASCADE HELIUM STORAGE - GAS GENERATOR HEATING

This cascade system (Figure 7) includes high temperature heat exchangers to decrease the helium mass requirements of the propellant tanks and therefore of the primary storage container. Heat sources are provided by gas generators. The heat exchanger located downstream of the primary container serves the same purpose as described for system 3, i.e., it reduces flowing pressurant density and therefore propellant tank pressurant mass requirements. The upstream heat exchanger (located between the primary and secondary helium storage containers) allows the secondary pressurant to be stored cryogenically, then expanded into the primary container at a high temperature. This technique should decrease primary storage container residuals, and at the same time decrease the size and mass of the secondary storage container.

H. SYSTEM 7 - MAIN TANK INJECTION - AMBIENT STORED HELIUM

The main tank injection (MTI) system, shown in Figure 8, represents one of the most advanced methods used in propellant tank pressurization. MTI is the process of generating pressurant gas within the confines of a propellant tank by injecting a reagent into the tank which reacts hypergolically with the propellant. Pressure control is accomplished by controlling the rate of reagent injection. The reagent is stored as a liquid at a pressure only slightly above propellant tank pressure. A very small, high pressure helium bottle

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FIGURE 7 CASCADE-ACTIVE FLOW HEATING

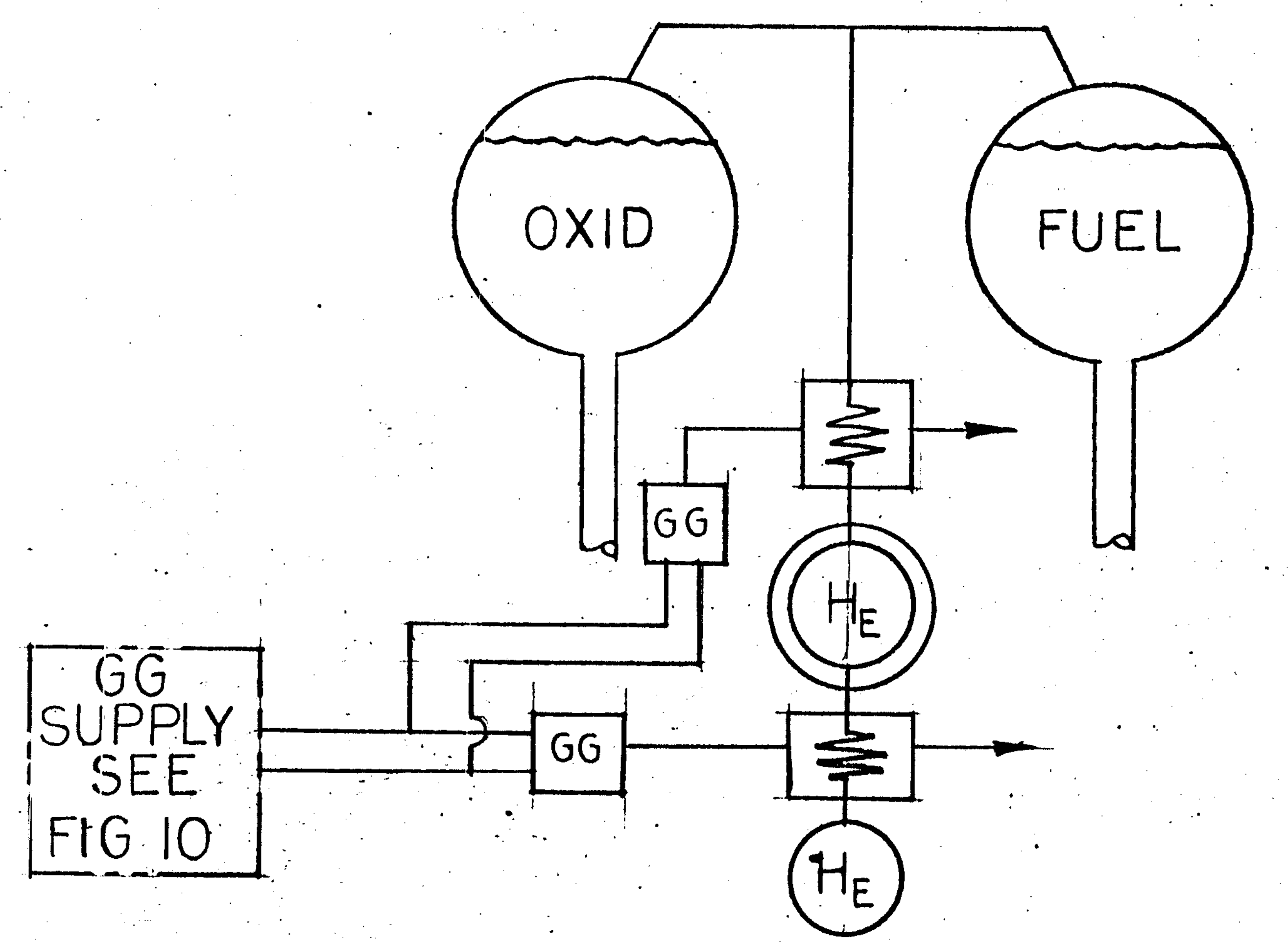
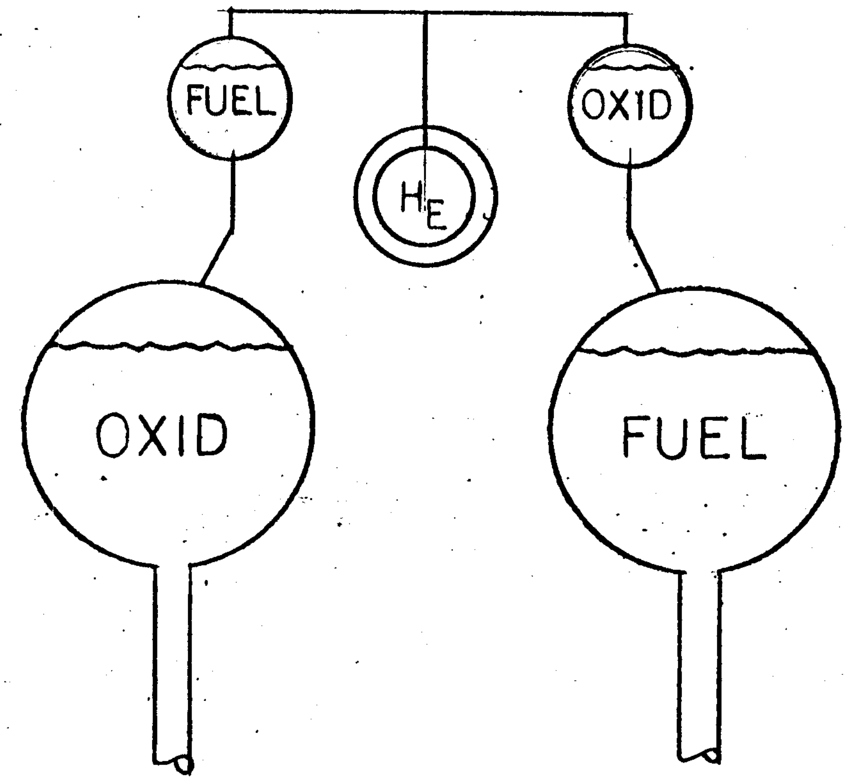


FIGURE 8 MAIN TANK INJECTION

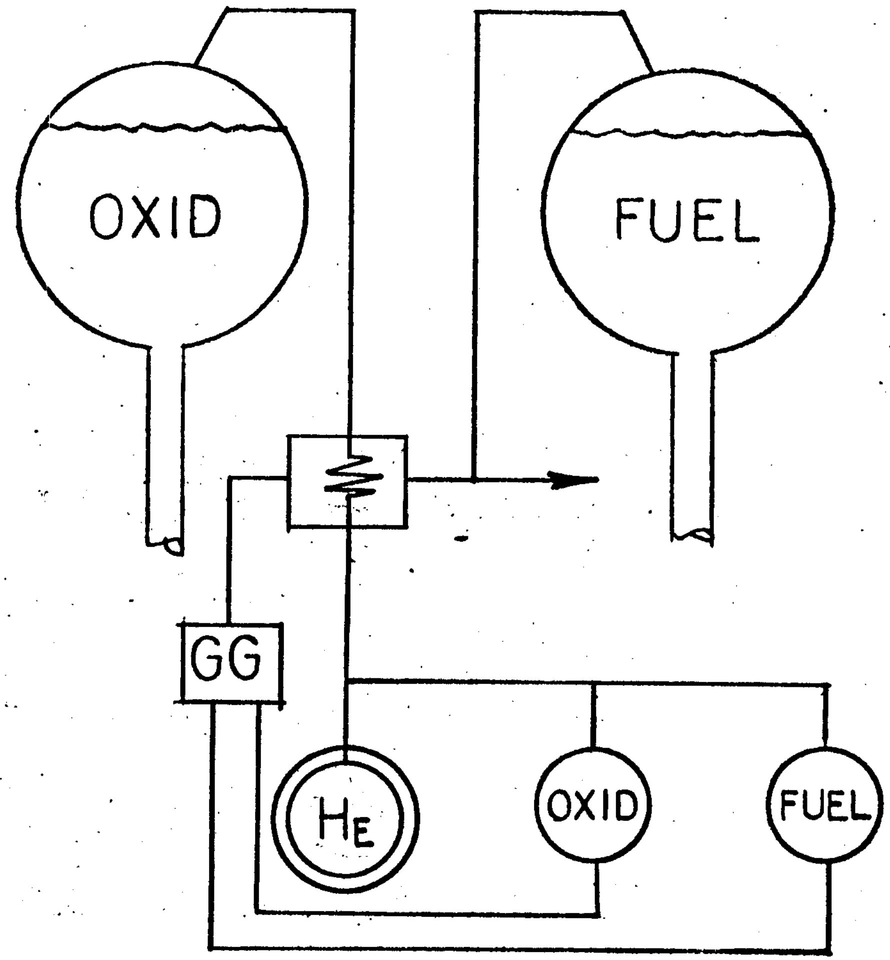


can be used for reagent pressurization. The high density, low pressure liquid reagent storage system represents a small fraction of the weight and volume of a gaseous helium pressurant storage container. Development of MTI systems for N_2O_4/Aerozine-50 propellants has been continuing at Martin-Denver for approximately two years. Feasibility of concept and operation has been demonstrated using full scale, flight weight tankage and hardware. System response is very high, even after extended shutdown periods. Ullage gas temperatures are relatively high, near 200°F, and the combustion product molecular weights are near 21 lb_m/lb-mole for the A-50 tank and 30 lb_m/lb-mole for the N_2O_4 tank.

I. SYSTEM 8 - CRYOGENIC STORED - HEATED HELIUM/GAS GENERATOR PRODUCTS

This system (Figure 9) is a variation of System 3; the only difference being that in System 8, the fuel rich gas generator combustion products are used to pressurize the fuel tank after serving as a helium gas heat source. This provides a "free" source of pressurant for the fuel tank, thereby decreasing the total system helium usage. It is not practical to consider the fuel rich gas generator products as a pressurant for the oxidizer tank because a gas generator using N_2O_4/A-50 must be operated at a very low oxidizer/fuel mixture ratio for stability and temperature reasons, and the resulting combustion products are reactive with the N_2O_4. Gas generator combustion products, having a molecular weight of 16 lb_m/lb-mole from a gas generator mixture ratio of .085, are used for fuel tank pressurization in the Titan II ICBM, Gemini Launch Vehicle, and Titan III Core Vehicle.

FIGURE 9 GAS GENERATOR



J. GAS GENERATOR PROPELLANT SUPPLY SUBSYSTEM

An auxiliary propellant supply system which can be used for most pressurization systems using a gas generator is shown in Figure 10. The gas generator will use N_2O_4 and .5 N_2H_4 - .5 UDMH as propellants. The two positive displacement accumulators shown are used for gas generator operation when the main engine is not operating (such as pre-start tank pressurization). The accumulators will be automatically refilled and gas generator bootstrap operation initiated when feed line propellant flow to the engine is initiated. Helium actuation pressure can be provided by the main pressurization helium supply. It should be noted that if the gas generator reaction products are used as a propellant tank pressurant, such as in System 8, feed line propellant bleed to the gas generator is not feasibile due to system pressure characteristics. In this case, the accumulators would have to be designed to supply propellants for the duration of the longest burn.

Upon completion of the survey, systems 1, 2, 4, 5, 7, and 8 were chosen as candidate systems to be subjected to the preliminary analysis and evaluation effort. An additional system, designated as 1A, was also included as a candidate system. System 1A, shown in Figure 11, is similar to system 1, but uses hydrogen to pressurize the fuel tank. Hydrogen was not considered as an oxidizer tank pressurant, due to the potential explosion hazard of a hydrogen/nitrogen tetroxide vapor mixture.

System 3 was deleted because of the similarity to system 4. System 6 was deleted due to complexity, and similarity in concept to system 5.

FIGURE 10 G.G. SUPPLY SYSTEM

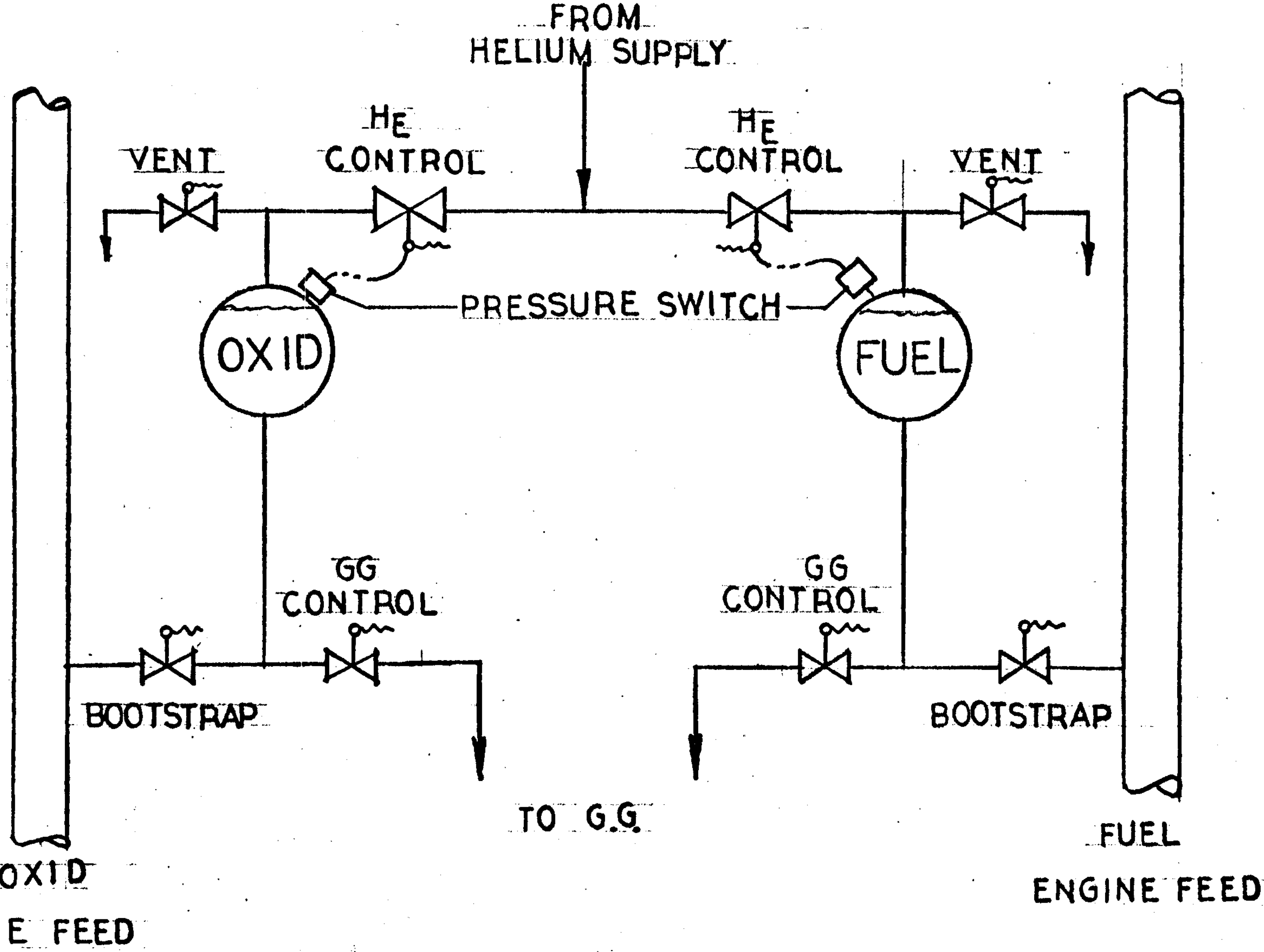
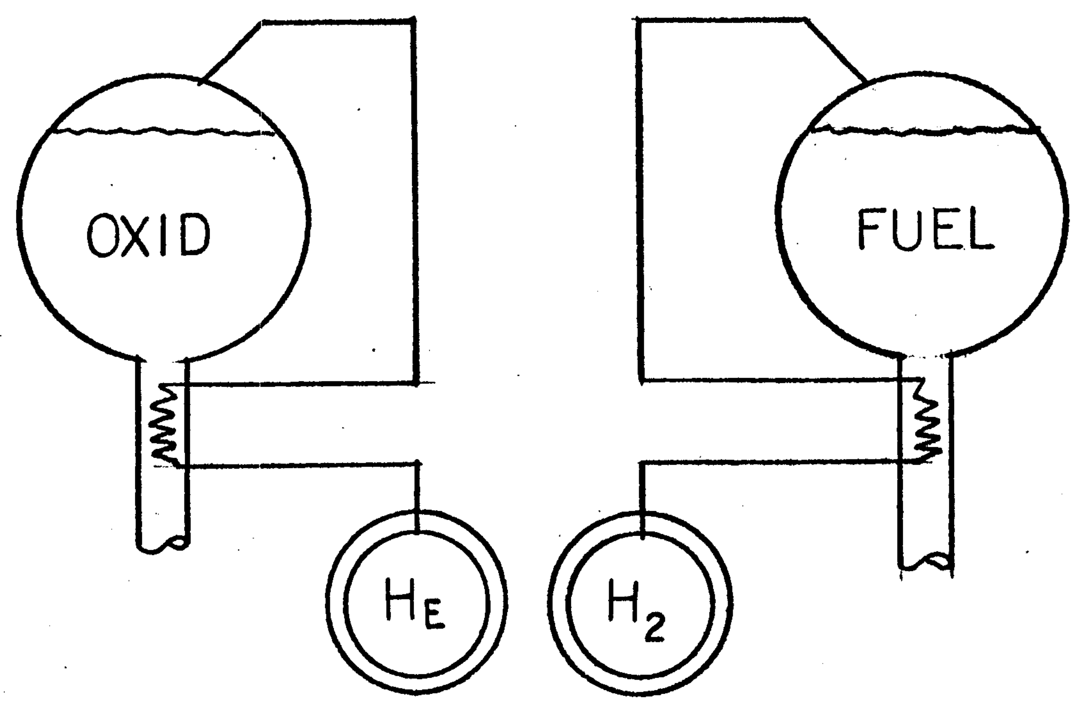


FIGURE II STORED HELIUM/HYDROGEN



III. PRELIMINARY STUDY OF PRESSURIZATION SYSTEMS

Subsequent to the completion of the pressurization system survey, the selected candidate systems were subjected to preliminary design and analysis. The purpose of this effort was to provide a basis for comparative evaluation of the candidate systems, so that the candidates showing the least potential could be omitted from more detailed investigation. The present Apollo SPS pressurization system was also subjected to the preliminary analysis, to provide a baseline reference for the comparison.

The initial effort of the preliminary study was to expand the basic candidate system schematics (Figures 2 - 9) to include all components required for proper operation of the systems. The final results of this functional design effort are shown in Figures 12 - 18. It should be noted that all valves, pressure switches, and check valves (with the exception of fill valves and vent valves) shown are considered as series-parallel redundant units, for the purpose of increased reliability.

Preliminary reliability estimates were established for each candidate system, and for the present Apollo SPS pressurization system. Each system was analyzed on the basis of the 215.92 hour mission duty cycle (Table 1), using established generic failure rate data for each type of component. Items such as fill disconnects, filters, and vent-relief valves were excluded from this study; those components are required for all the candidate systems, and therefore, do not contribute to a comparative evaluation of the candidates. Components which did enter into the reliability studies were flow control valves, check valves, pressure switches, heat exchangers, gas generators, pressurant storage containers, lines

(1) N_2O_4 TANK VENT VALVE	(8) GH_e FILL & DRAIN PRESS. SW.	(15) A-50 VENT PRESS. SW.
(2) N_2O_4 VENT PRESS. SW.	(9) GH_e TANK	(16) A-50 TANK
(3) N_2O_4 TANK	(10) PRESS. CONTROL VALVE	(17) GH_e HEAT EXCHANGER.
(4) GH_e HEAT EXCHANGER	(11) PRESS. CONTROL SW.	(18) A-50 TANK F.& D. VALVE
(5) N_2O_4 TANK F.& D. VALVE	(12) — DELETED —	(19) A-50 PRE VALVE
(6) N_2O_4 PRE VALVE	(13) — DELETED —	(20) G He TANK VENT VALVE
(7) GH_e TANK F.& D. VALVE	(14) A-50 TANK VENT VALVE	(21) G He VENT PRESS. SW.

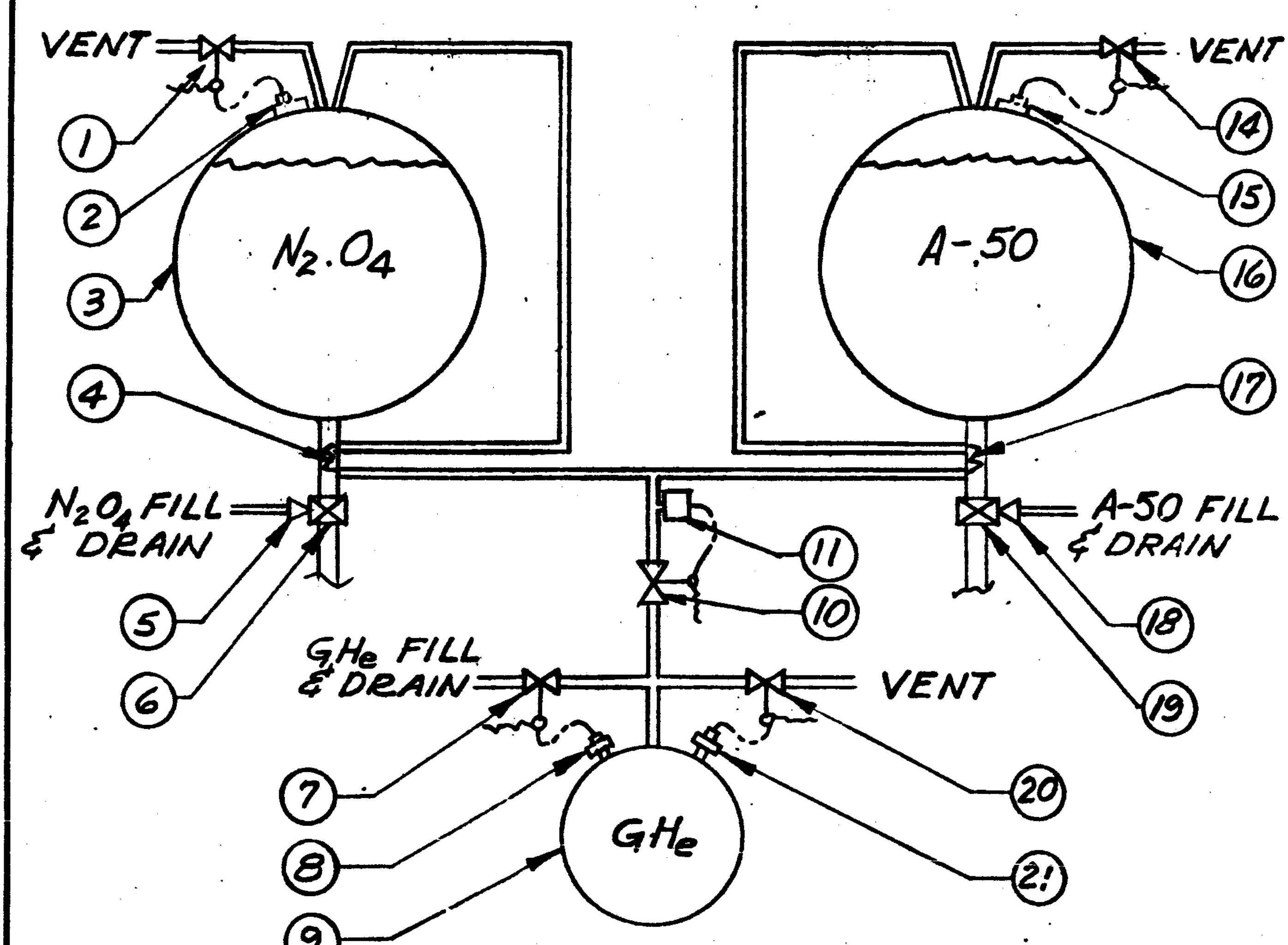


FIGURE 12

TITLE: ADV. LT.-WT. PRESSURIZATION SYS. — STORED HELIUM SYSTEM — PASSIVE FLOW HEATING

SYS NUMBER 1

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(1) CHECK VALVE	(10) GH_e TANK	(19) PRESS. SWITCH - H_e VENT
(2) VENT VALVE - OXID TANK	(11) CHECK VALVE	(20) PRESS. SWITCH - H_2 VENT
(3) PRESS. SWITCH - OXID TANK VENT	(12) PRESS. SWITCH - H_e CONTROL	(21) GH_2 TANK
(4) OXIDIZER TANK	(13) H_e CONTROL VALVE	(22) VENT VALVE - FUEL TANK
(5) HEAT EXCHANGER	(14) H_e CONTROL VALVE	(23) PRESS. SWITCH - FUEL TANK VENT
(6) H_e F&D VALVE	(15) PRESS. SWITCH - H_e CONTROL	(24) HEAT EXCHANGER
(7) OXIDIZER PRE-VALVE	(16) FUEL TANK	(25) H_2 F&D VALVE
(8) OXIDIZER F&D VALVE	(17) VENT VALVE - H_e TANK	(26) FUEL PRE-VALVE
(9) PRESS. SWITCH - H_e F&D	(18) VENT VALVE - H_2 TANK	(27) FUEL F&D VALVE
		(28) PRESS. SWITCH - H_2 FILL

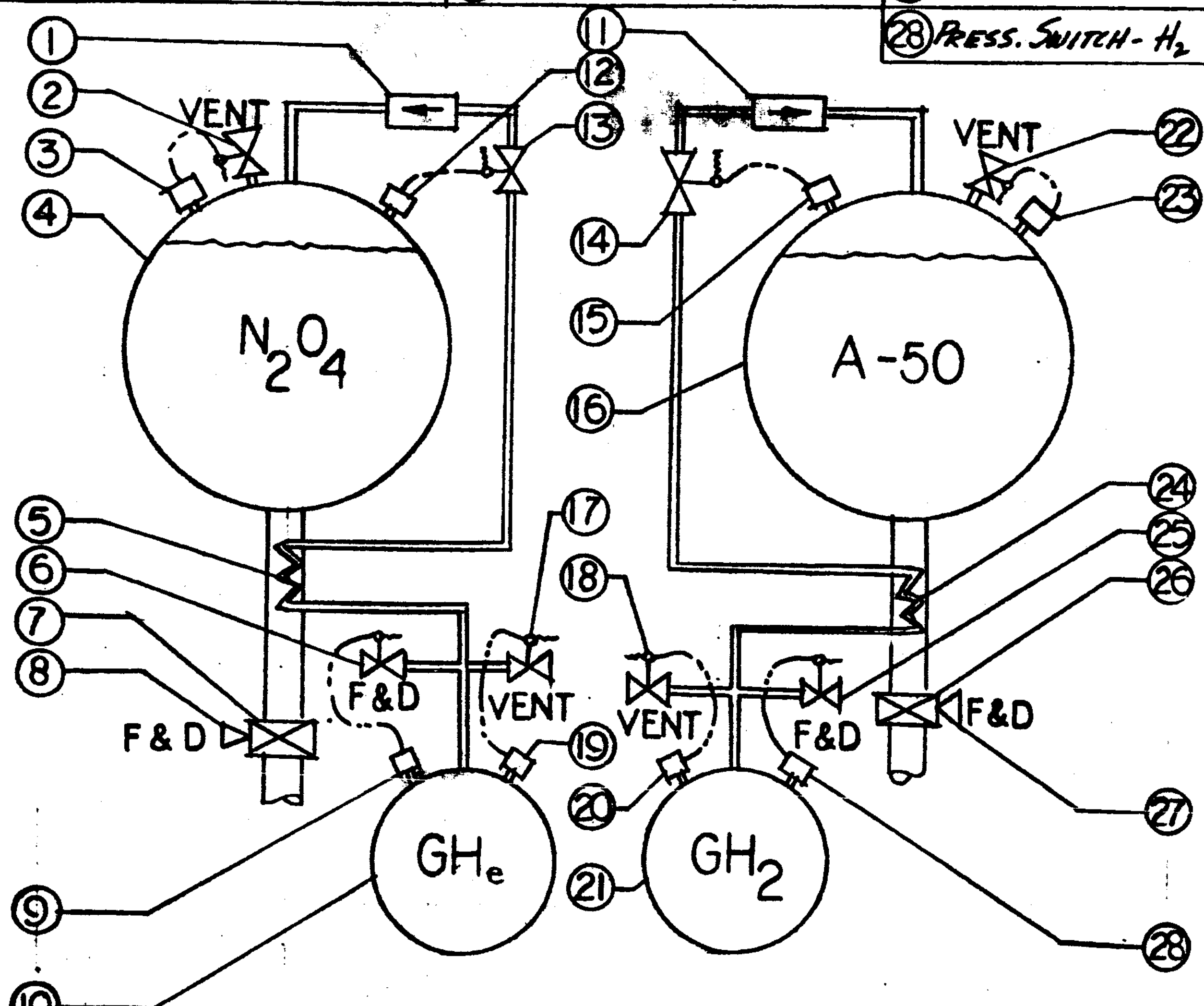

FIGURE 13

TITLE: ADVANCED LIGHT-WEIGHT PRESS. SYSTEM - STORED HELIUM AND HYDROGEN - PASSIVE FLOW HEATING	SYS. NUMBER 1 A
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PAGE	CHG.

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① OXID TANK VENT VALVE	⑫ He F&D VALVE	㉓ FUEL PRE-VALVE
② OXID TANK VENT PRESS. SWITCH	⑬ PRESS. SWITCH - He FILL	㉔ FUEL F&D VALVE
③ OXIDIZER TANK	⑭ CHECK VALVE	㉕ G.G. FUEL VALVE
④ PRESS. SWITCH - He CONTROL	⑮ HELIUM CONTROL VALVE	㉖
⑤ OXIDIZER PRE-VALVE	⑯ VENT VALVE - He TANK	㉗
⑥ OXIDIZER F&D VALVE	⑰ PRESS. SWITCH - He VENT	㉘
⑦ GHe TANK	⑱ HEAT EXCHANGER	
⑧ GAS GENERATOR	⑲ PRESS SWITCH - FUEL TANK VENT	
⑨ G.G. OXIDIZER VALVE	⑳ FUEL TANK VENT VALVE	
⑩ HEAT EXCHANGER	㉑ FUEL TANK	
⑪ CHECK VALVE	㉒ HEAT EXCHANGER	

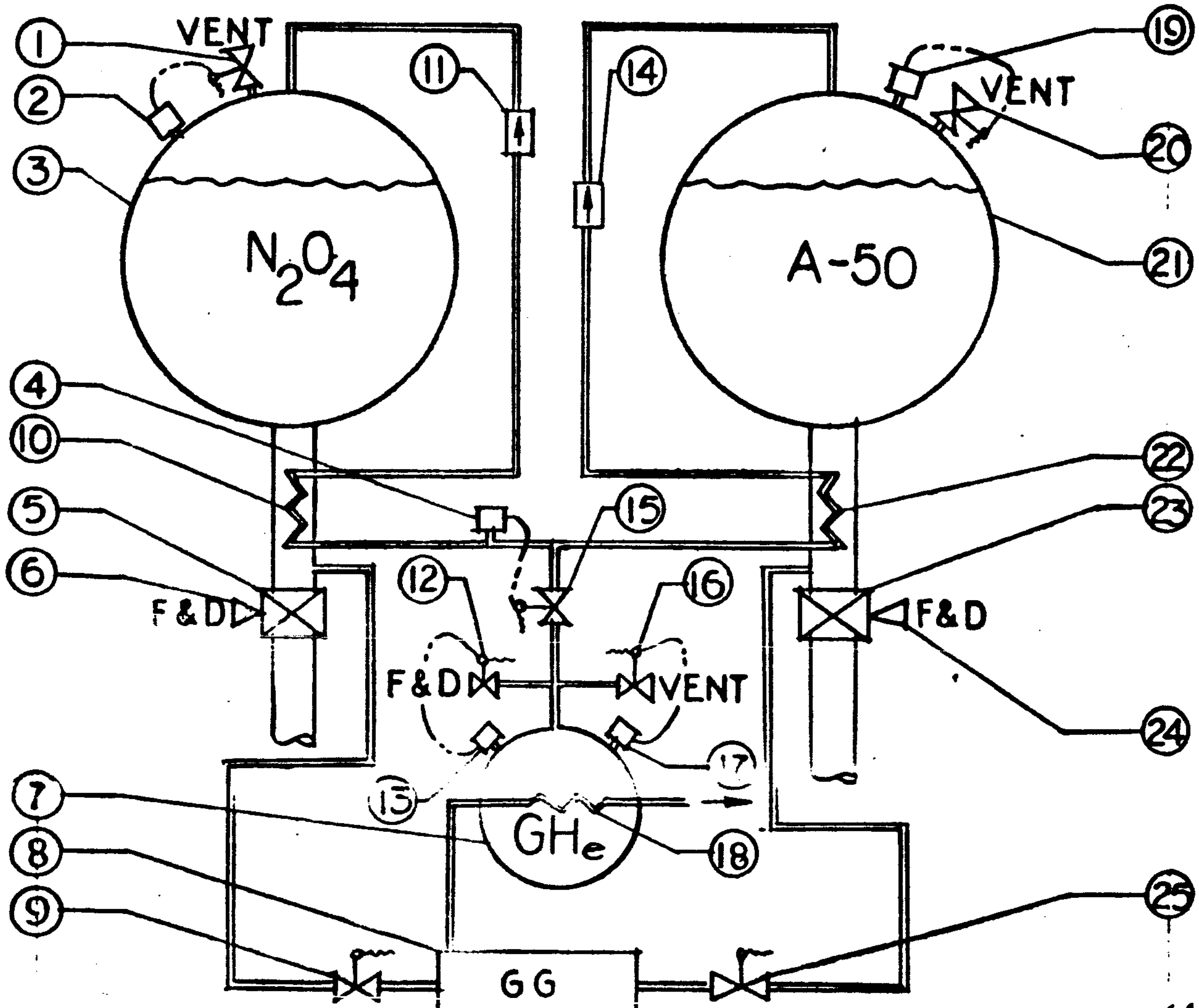

FIGURE 1A

TITLE: ADV. LT-WT. PRESSURIZATION SYSTEM
STORED HELIUM SYS. - PASSIVE
FLOW HEATING - RESIDUAL HEATING

SYS NUMBER 2

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(1) N_2O_4 TANK VENT VALVE	(8) G.G. OXIDIZER VALVE	(15) A-50 TANK
(2) N_2O_4 VENT PRESS. SW.	(9) CHECK VALVE	(16) A-50 PRE-VALVE
(3) N_2O_4 TANK	(10) He HEAT EXCHANGER	(17) A-50 FILL & DRAIN VAL.
(4) N_2O_4 PRE VALVE	(11) CHECK VALVE	(18) He TANK
(5) N_2O_4 FILL & DRAIN VALVE	(12) PRESS. SWITCH-He CONT.	(19) He TANK FILL VALVE
(6) He SOLENOID VALVE	(13) A-50 TANK VENT VALVE	(20) He TANK VENT VALVE
(7) GAS GENERATOR	(14) A-50 VENT PRESS. SW.	(21) G.G. FUEL VALVE
		(22) PRESS. SWITCH - He VENT
		(23) PRESS. SWITCH - He F & D

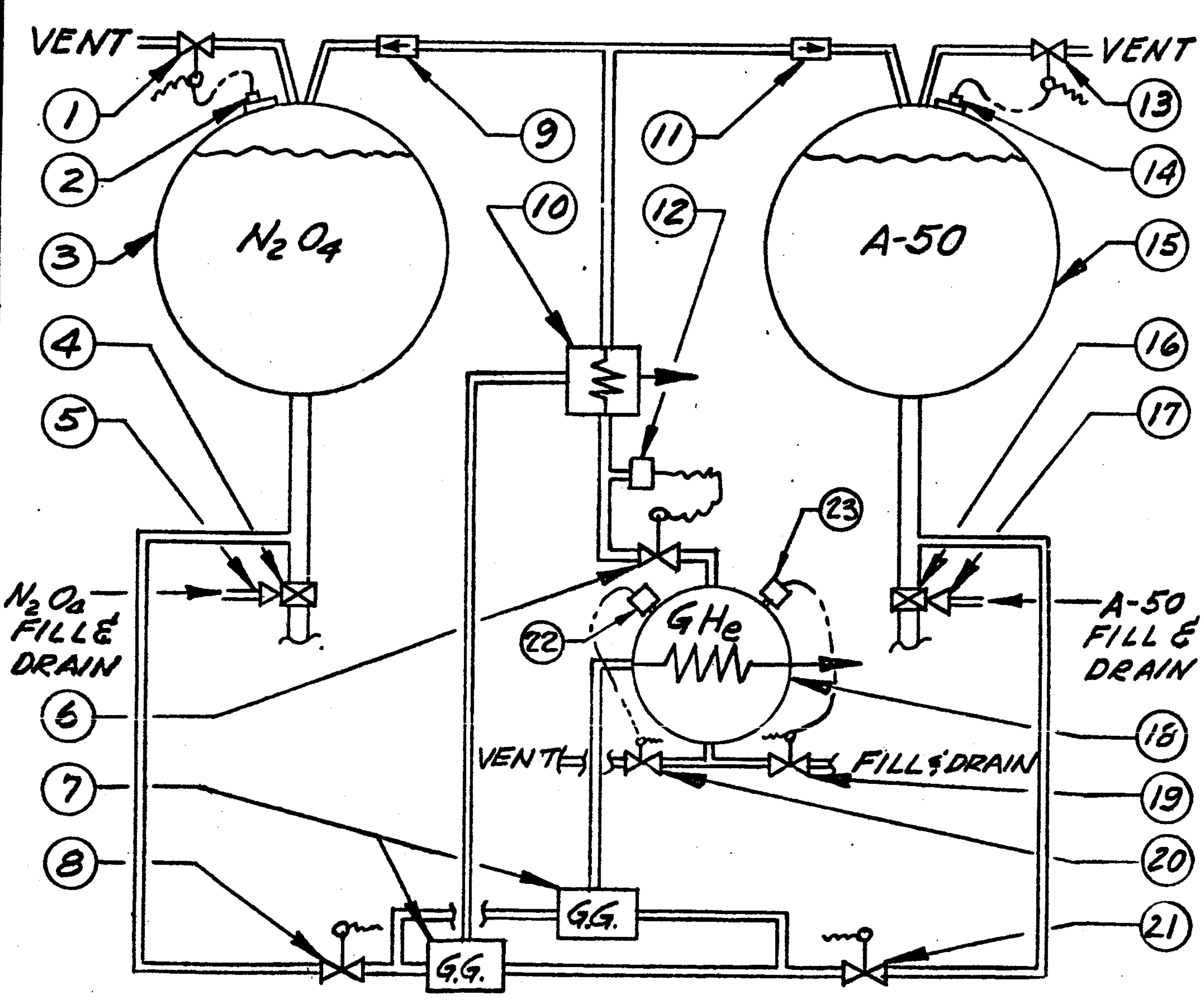

FIGURE 15

TITLE: ADV. LT-WT PRESSURIZATION SYS. STORED HELIUM SYSTEM - ACTIVE FLOW HEATING - RESIDUAL HEATING	SYS NUMBER 4		
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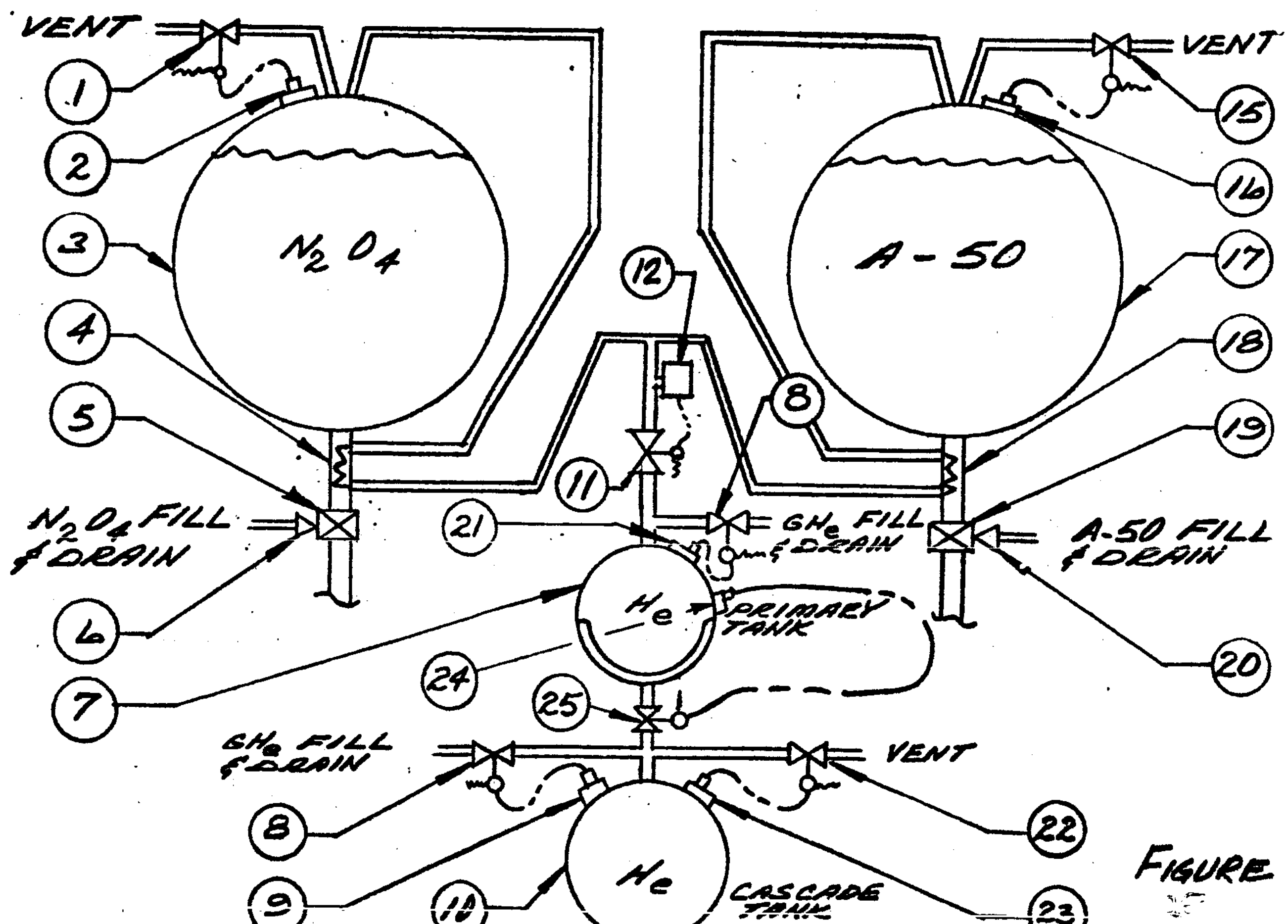
1 N_2O_4 TANK VENT VALVE	10 GHe TANK	19 A-50 PRE VALVE
2 N_2O_4 VENT PRESS. SW.	11 PRESS. CONTROL VALVE	20 A-50 TANK FILL & DRAIN VALVE
3 N_2O_4 TANK	12 PRESS. CONTROL SWITCH	21 GHe FILL & DRAIN SW.
4 GHe HEAT EXCHANGER	13 — DELETED —	22 GHe TANK VENT VALVE
5 N_2O_4 PRE VALVE	14 — DELETED —	23 GHe VENT PRESS. SW.
6 N_2O_4 FILL & DRAIN VALVE	15 A-50 TANK VENT VALVE	24 GHe PRESS. SW.
7 GHe ACCUMULATOR	16 A-50 VENT PRESS. SW.	25 GHe SHUTOFF VALVE
8 GHe TANK FILL & DRAIN VALVE	17 A-50 TANK	
9 GHe FILL & DRAIN PRESS. SW.	18 GHe HEAT EXCHANGER	



FIGURE

TITLE: ADVANCED LIGHT-WEIGHT PRESS. SYSTEM - CASCADE HELIUM STORAGE SYSTEM WITH PASSIVE FLOW HEATING.

SYS. NUMBER 5

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APPROVED: H. Forman

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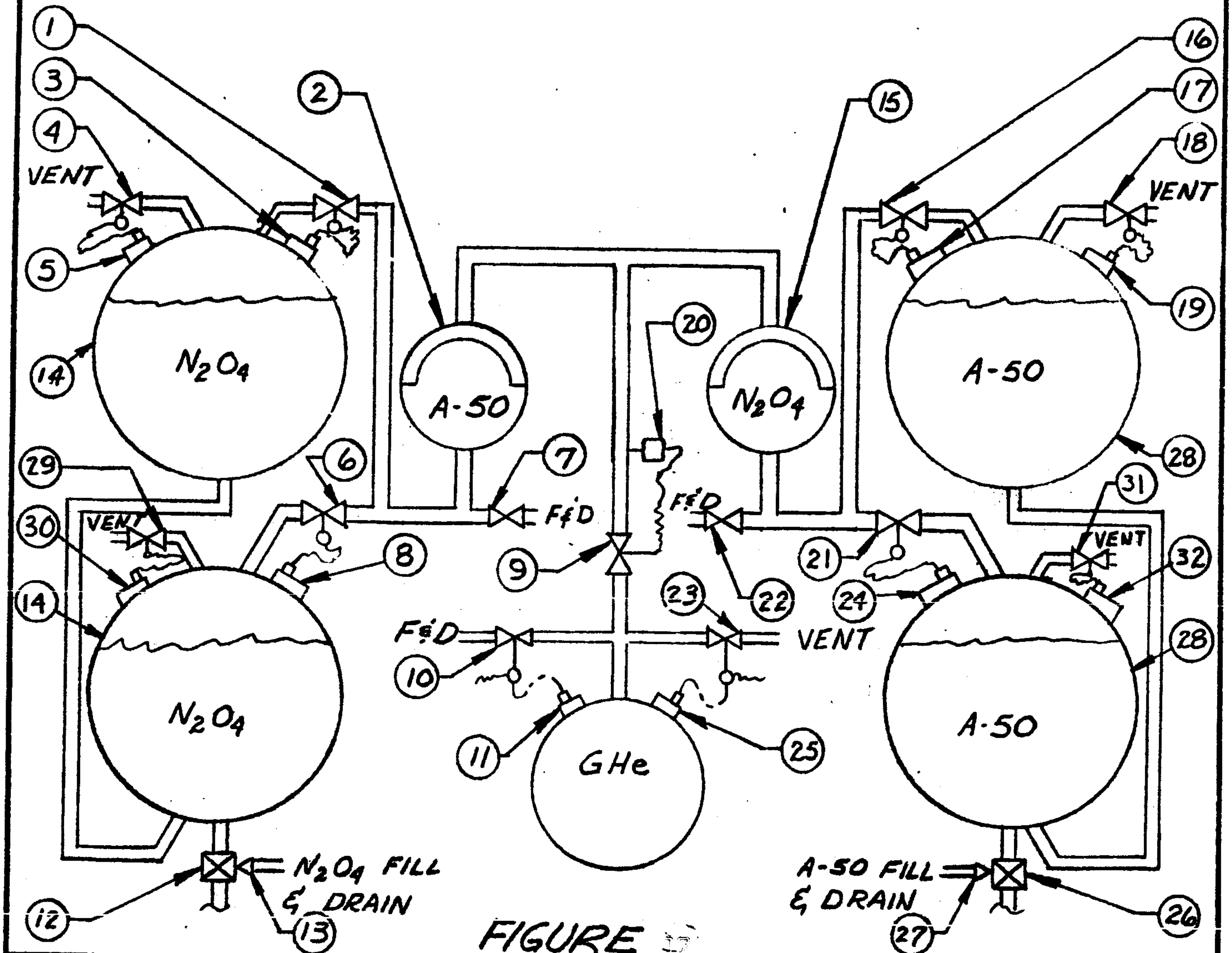
(1) N_2O_4 PRESS CONT VALVE	(12) N_2O_4 PRE VALVE	(23) GHe VENT VALVE
(2) A-50 ACCUMULATOR	(13) N_2O_4 FILL & DRAIN VALVE	(24) A-50 PRESS CONT SW
(3) N_2O_4 PRESS CONT. SW	(14) N_2O_4 TANK	(25) GHe VENT PRESS SW
(4) N_2O_4 TANK VENT VAL.	(15) N_2O_4 ACCUMULATOR	(26) A-50 PRE VALVE
(5) N_2O_4 VENT PRESS. SW.	(16) A-50 PRESS CONT VALVE	(27) A-50 FILL & DRAIN VAL
(6) N_2O_4 PRESS CONT VALVE	(17) A-50 PRESS CONT SW	(28) A-50 TANK
(7) A-50 FILL & DRAIN VAL	(18) A-50 TANK VENT VAL	(29) N_2O_4 TANK VENT VAL
(8) N_2O_4 PRESS CONT SW	(19) A-50 VENT PRESS SW	(30) N_2O_4 VENT PRESS. SW
(9) GHe PRESS CONT VAL	(20) GHe PRESS SW	(31) A-50 TANK VENT VAL
(10) GHe FILL & DRAIN VAL	(21) A-50 PRESS CONT VALVE	(32) A-50 VENT PRESS SW
(11) GHe FILL & DRAIN PRESS SW	(22) N_2O_4 FILL & DRAIN VAL	



FIGURE

TITLE: ADV. LT-WT. PRESSURIZATION SYS- MAIN TANK INJECTION SYS.

SYS. NUMBER 7

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(1) N_2O_4 TANK VENT VALVE	(9) GHe FILL & DRAIN PRESS. SW.	(17) A-50 TANK
(2) N_2O_4 VENT PRESS. SW.	(10) GHe TANK	(18) A-50 PRE VALVE
(3) N_2O_4 TANK	(11) N_2O_4 PRESS. CONT. VALVE	(19) A-50 TANK F. & D. VALVE
(4) N_2O_4 PRE VALVE	(12) N_2O_4 PRESS. CONT. SW.	(20) GH_e HEAT EXCHANGER
(5) N_2O_4 TANK F. & D. VALVE	(13) A-50 PRESS. CONT. VALVE	(21) A-50 ACCUMULATOR
(6) GAS GENERATOR	(14) A-50 PRESS. CONT. SW.	(22) ACCUM. A-50 VALVE
(7) N_2O_4 ACCUMULATOR	(15) A-50 TANK VENT VALVE	(23) GH_e TANK VENT VALVE
(8) GH_e TANK F. & D. VALVE	(16) A-50 VENT PRESS. SW.	(24) GH_e VENT PRESS. SW.
	(26) G.G. RELIEF VALVE	(25) ACCUM. N_2O_4 VALVE

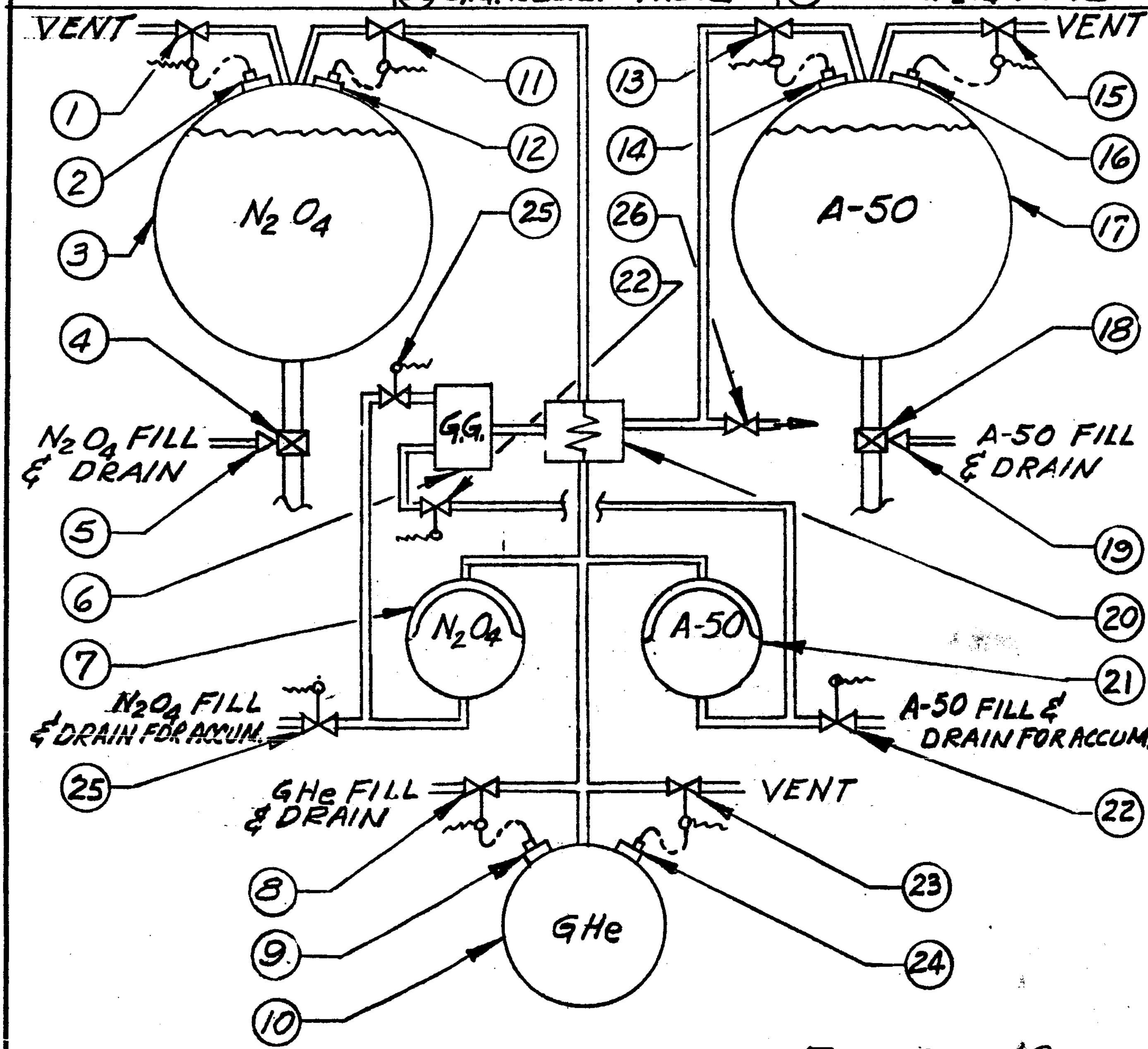

FIGURE 18

TITLE: ADV. LT.-WT. PRESS. SYSTEM — STORED HELIUM-N_2O_4 TANK-ACTIVE FLOW HEATING-AUTOGENOUS G.G. PRODUCTS-A-50 TANK	SYS. NUMBER 8

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Table 1 - Advanced Lightweight Pressurization
System Design Mission Data

EVENT	TIME FROM LIFT OFF	DURATION
Launch	00.00 Hrs.	
Earth Orbit Injection	.20	
Translunar Orbit Injection	4.70	
First Translunar Midcourse Correction	22.79	13.00 Sec
Second Translunar Midcourse Correction	40.79	13.00
Third Translunar Midcourse Correction	58.79	13.00
Lunar Orbit Insertion	76.79	390.2
First Lunar Orbit Plane Change	80.88	20.20
Second Lunar Orbit Plane Change	127.38	10.20
Trans-earth Orbit Injection	133.38	121.0
First Trans-earth Midcourse Correction	160.92	3.20
Second Trans-earth Midcourse Correction	188.42	3.20
Thrid Trans-earth Midcourse Correction	215.92	3.20
	Total	590.2 Sec

Prelaunch Hold Time = 10.0 hours

Propellant Flow Rate, Total = 68.69 lbs/sec

Mixture Ratio (Nominal) = 2.00

$\dot{W}_{ox}$ = 45.79 lbs/sec $\dot{W}_f$ = 22.90 lbs/sec

Propellant Tankage:	Oxidizer (2)	Fuel (2)
Total Volume (maximum)	321. ft^3	255.6 ft^3
Ullage (minimum)	12.2	9.3
Operating Pressure (nominal)	175 psia	175 psia
Propellant Temperature (lims.)	40-80°F	40-80°F

and fittings. The resulting reliability numbers are tabulated in Table 2.

The preliminary system sizing and mass analysis was performed in accordance with the following stipulations:

1) Thermal effects of the environment were neglected - outer surfaces of pressurant and propellant storage tanks were adiabatic.

2) Subsystem sizing was established on the basis of nominal vehicle requirements only and did not provide for helium usage margin.

3) All analyses were based upon the design mission profile defined in Table 1.

4) The mass of tubing, insulation, support structure, and miscellaneous fittings were omitted. The mass analysis considered pressurant, pressurant storage containers, valves, pressure switches, gas generators (and propellants), and heat exhcangers.

The present Apollo pressurization system was also analyzed using the above ground-rules. Rather than use the actual mass, the Apollo system was "re-weighed" to reflect the same criteria and techniques used in analyzing the candidate systems. In this way, all systems were compared in the proper relative perspective.

Table 2 - Candidate System Reliability Comparison

System	Reliability
Present Apollo	.999451
1	.999374
1A	.999373
2	.999142
4	.999076
5	.999181
7	.999284
8	.999268

The analysis and optimization of the candidate system entailed parametization about three major variables: temperature of pressurant entering the propellant tanks, initial pressurant storage temperature, and initial pressurant storage pressure.

A. SYSTEMS 0, 1, and 1A - PROPELLANT TANK PRESSURANT USAGE

Since the mass of helium required to pressurize the propellant tanks is a function only of the temperature of the helium entering the tanks and the conditions within the tanks, it was convenient to make this analysis for all helium systems at one time. The mass of hydrogen required for fuel tank pressurization in system 1A was also determined at this time. An IBM 7094 computer program (Martin CR-65-10, "Utilization Instructions - Tank Pressurization Computer Program ØD041," February 1965) was used to calculate the pressurant masses required for fuel and oxidizer tanks, for several pressurant entering temperatures. The results are plotted in Figures 19, 20, and 21. These figures also illustrate the effect of pressurant entering temperature upon total mass of propellants vaporized during the mission.

Considering the fuel tank requirements (Figures 19 and 20), it is noted that inlet temperature has almost no effect upon pressurant mass, particularly above 500°R. Also, the effect upon vaporized fuel is negligible, due to the comparatively low vapor pressure of the N_2H_4/UDMH mixture at operating temperatures. The oxidizer tank pressurant curve (Figure 21) also shows only a very slight decrease in pressurant at temperatures above 500°R. Oxidizer vapor mass is

FIGURE 19

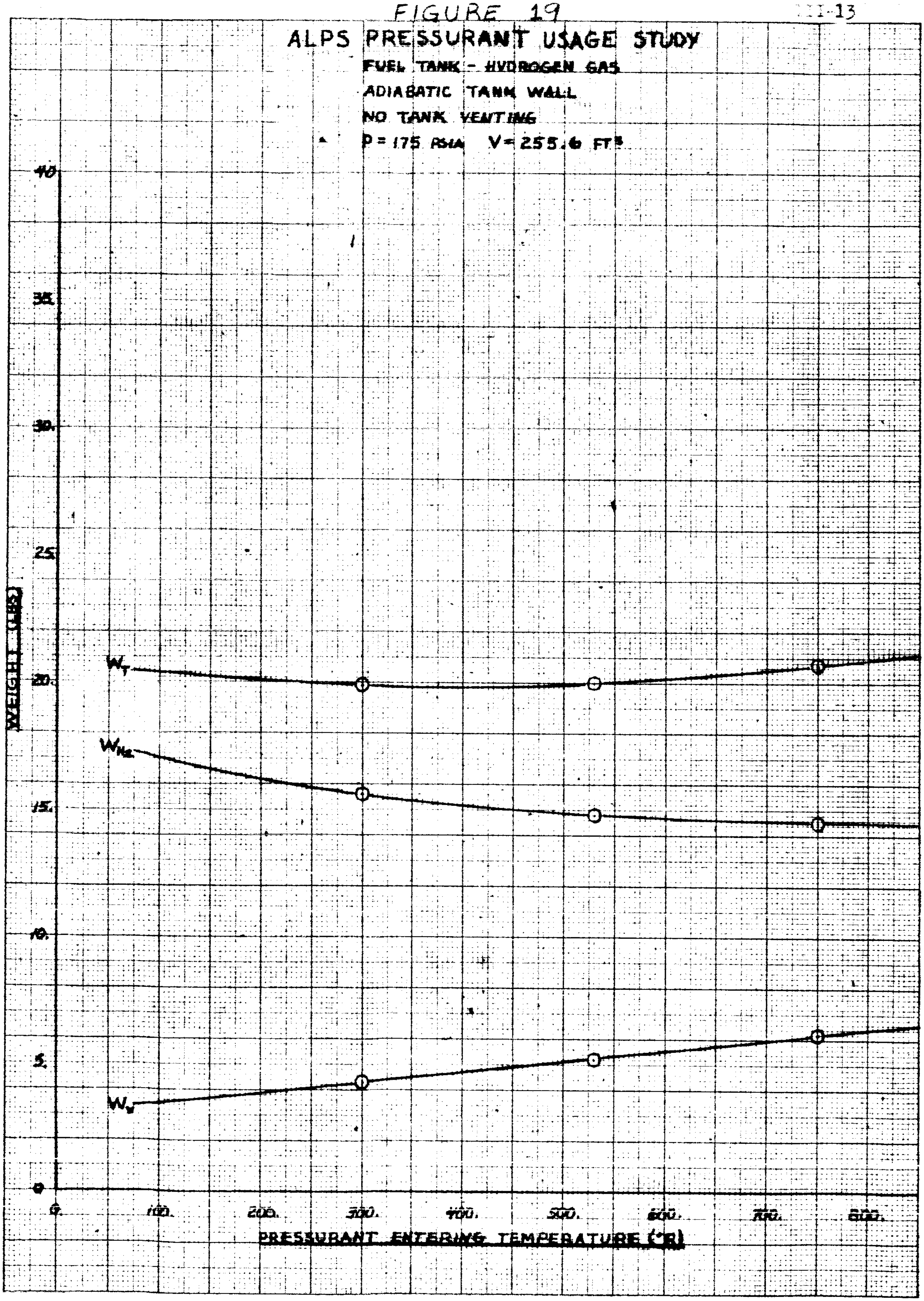

ALPS PRESSURANT USAGE STUDY
FUEL TANK - HYDROGEN GAS
ADIABATIC TANK WALL
NO TANK VENTING
P = 175 PSIA V = 255.6 FT³
WEIGHT (LBS)
40
35
30
25
20
15
10
5
0
W_T
W_H2
W_s
0
100
200
300
400
500
600
700
800
PRESSURANT ENTERING TEMPERATURE (°R)

FIGURE 20

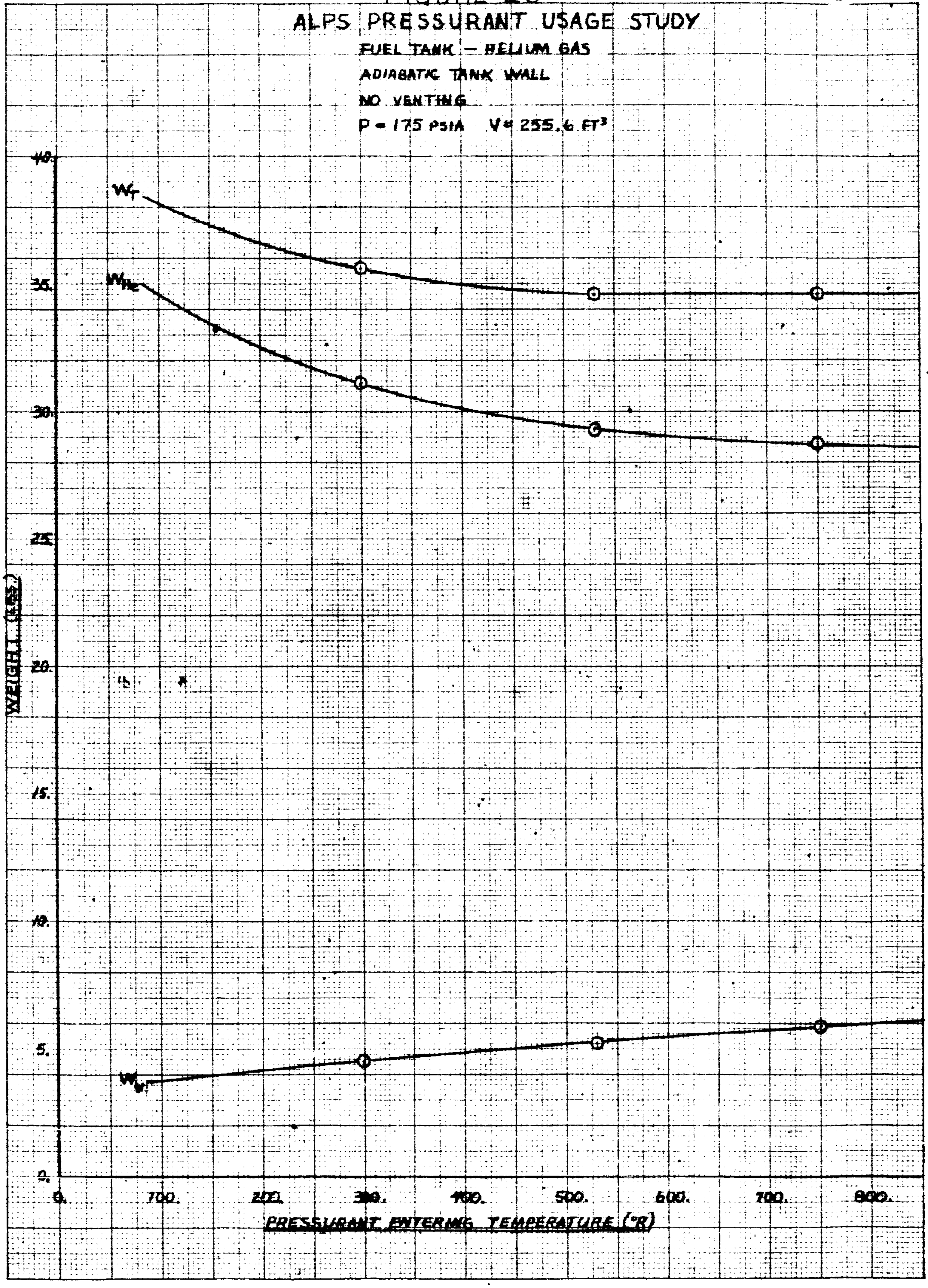
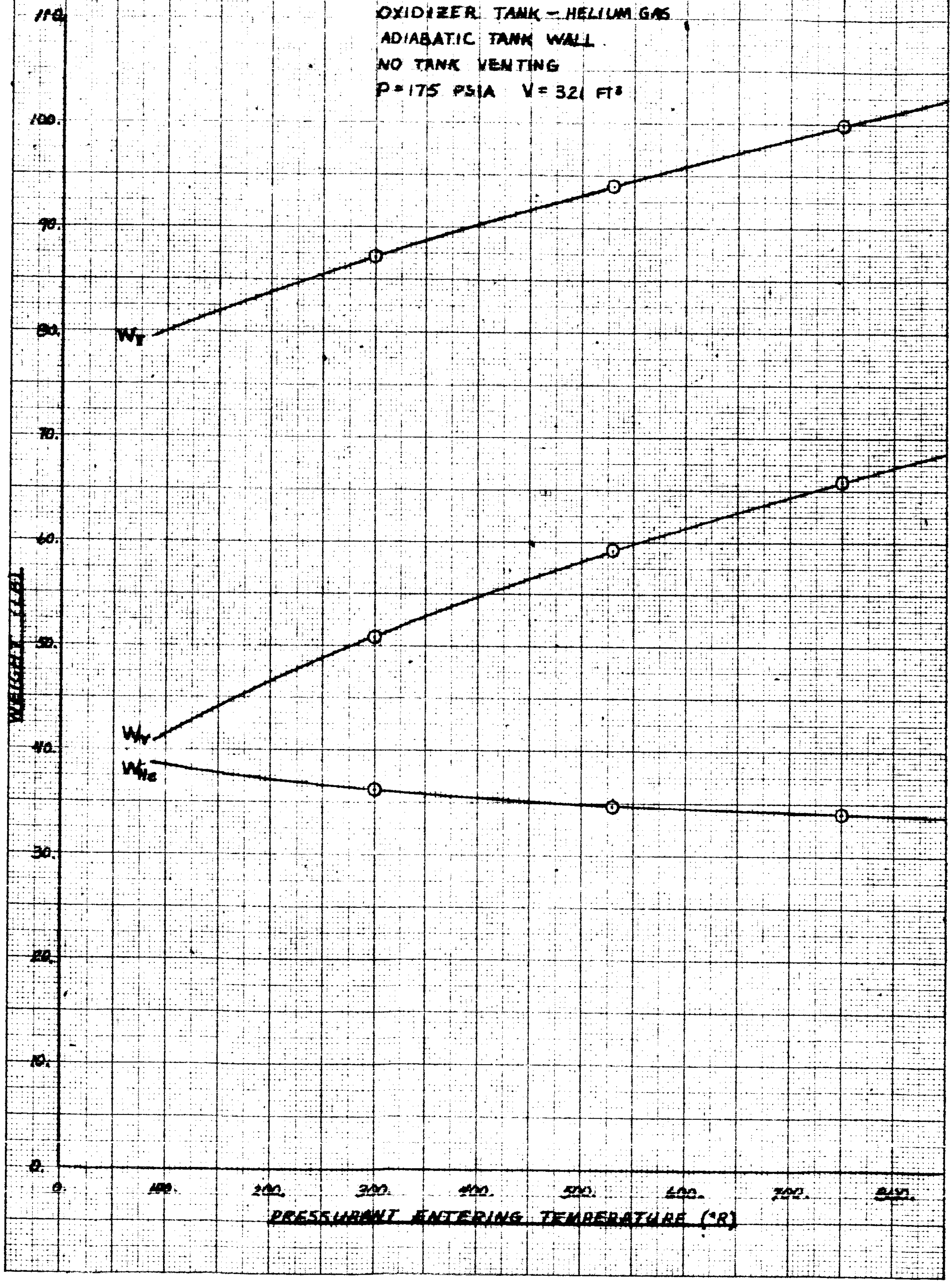

ALPS PRESSURANT USAGE STUDY
FUEL TANK — HELIUM GAS
ADIABATIC TANK WALL
NO VENTING
P = 175 PSIA V = 255.6 FT³
W_T
W_{He}
W_{W_T}
WEIGHT (LBS.)
0. 5. 10. 15. 20. 25. 30. 35. 40.
0. 100. 200. 300. 400. 500. 600. 700. 800.
PRESSURANT ENTERING TEMPERATURE (°R)

FIGURE 21

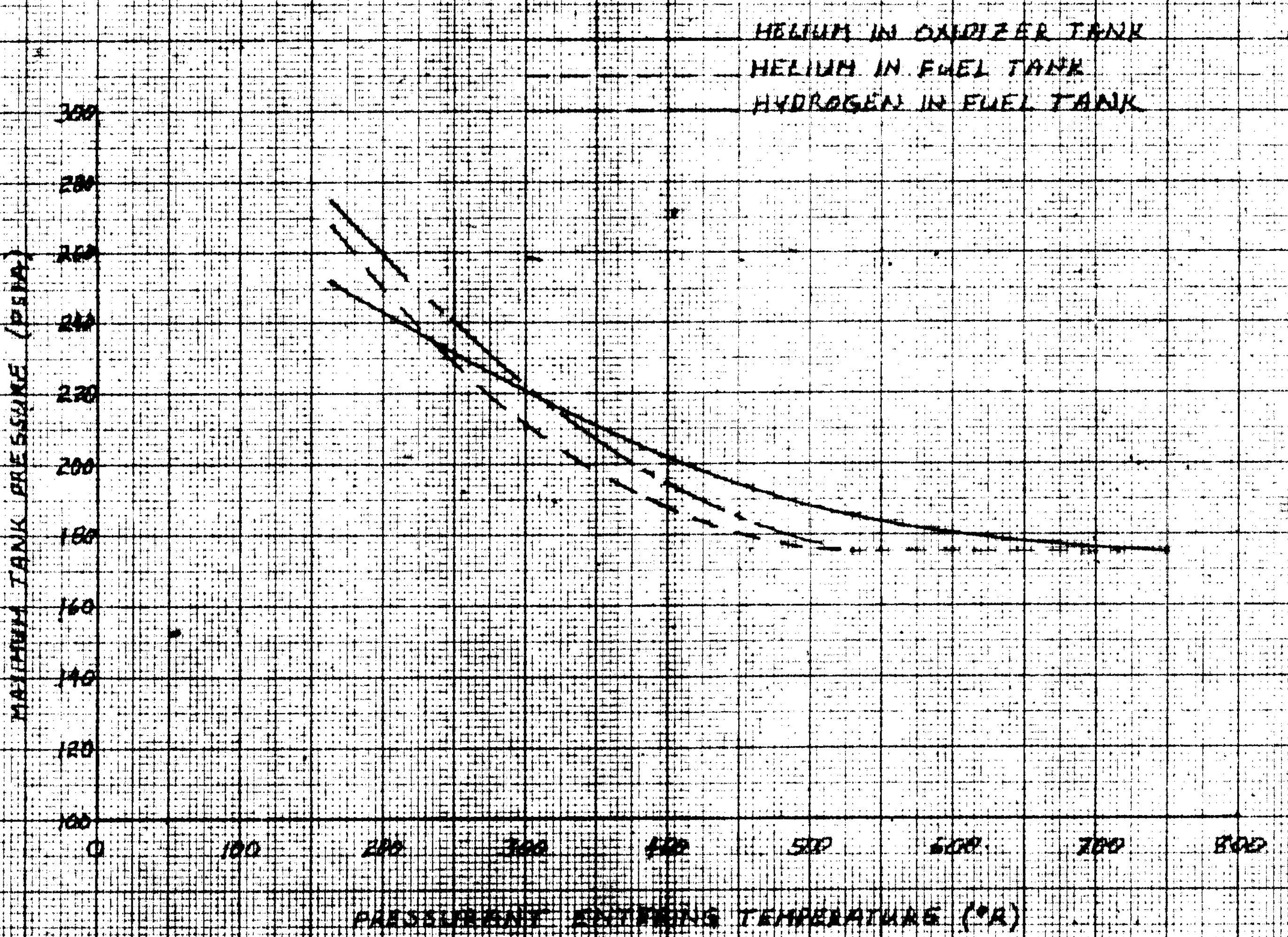
ALPS PRESSURANT USAGE STUDY



much more sensitive to temperature, but this factor is of minor importance when compared to total pressurization system mass. It was concluded that with a stored gas system for the Apollo Service Propulsion System, there is no advantage in heating the pressurant to a level significantly above the nominal propellant temperature. System weight reductions must be effected through changes in pressurant storage technique rather than by increasing the pressurant entering temperature.

There is, in addition to pressurant mass requirements, another effect of pressurant entering temperature which was noted during the preliminary analysis. This is the influence of pressurant temperature upon propellant storage tank pressure. With the onset of each vehicle coast period, the bulk propellants, ullage gases and propellant tank walls will tend to attain uniform thermal equilibrium. If, at the end of a burn period, the tank ullage temperature is above ambient, then the subsequent cooling will cause a decrease in propellant tank pressure. If the ullage temperature is below ambient at the end of the burn period, tank pressure will rise as equilibrium takes place. Figure 22 shows maximum tank pressures as a function of pressurant entering temperature. The data shown in this figure were taken from computer runs which analyzed the propellant tank thermodynamic histories over the design mission. The maximum operating pressure for the Apollo Service Propulsion System is 225 lb_f/in^2 absolute; therefore, entering gas temperatures less than above 300°R would cause propellant tank venting in the course of a normal mission.

FIGURE 4.2 MAXIMUM PROPELLANT TANK PRESSURE VS. PRESSURANT TEMPERATURE
ADIABATIC TANK WALLS



System mass is relatively unaffected by pressurant entering temperature, as long as the entering temperature is above 300°R. Therefore, ambient entering gas temperature was selected to minimize propellant tank pressure excursions. For purposes of this preliminary study, ambient temperature was taken as 530°R. Propellant tank pressurant usage was therefore fixed at 29.3 lb_m and 34.7 lb_m of helium for fuel and oxidizer tanks, respectively, and 14.8 lb_m of hydrogen for system 1A fuel tank.

Pressurant Storage

Storage container weight was optimized from the standpoint of initial pressure and initial temperature.[1] Initial storage pressures considered were 1000, 2000, 3000, and 4000 lb_f/in^2 absolute. In all cases, the final and/or minimum storage pressure was fixed at 400 lb_f/in^2 absolute. Initial helium storage temperatures investigated were 37, 140, 300, and 530°R. In the case of hydrogen storage, the temperatures considered were 70, 140, 300, and 530°R; the minimum temperature of 70°R was chosen to prevent the hydrogen from condensing in the storage container. Only spherical geometry was considered. Although the outer surface of the container was considered adiabatic, heat transfer between the sphere wall and the pressurant was considered. Also, the sphere wall and pressurant were forced to thermal equilibrium during each coast period.

A mathematical model was used to simulate the helium expansion process. This program is described in Martin CR-65-37, "Preliminary Utilization Instructions - Gas Expansion Computer Program,"

1. This discussion is pertinent to the analysis of all high pressure gas storage containers in all systems, with the exception of the primary storage tank in system 5.

May 1965. The program was originally written for helium gas, but was also suitable for hydrogen gas with a slight revision of the equation of state used. Also included in this program are the equations for calculating the storage container mass, based upon particular material properties, safety factors, and required dimensions. The spehere mass thus derived does not include any allowance for structural land areas or bosses which may be required.

It is illustrative to present the pressurant storage system mass data in terms of a dimensionless parameter called the "expansion ratio", which is defined as

$$\text{E.R.} = \frac{\text{Initial helium mass + Storage container mass}}{\text{Expelled helium mass}} .$$

This ratio is a constant for all similar expansion processes (i.e., adiabatic, isobaric, isothermal) which are not time-dependent in nature, and is independent of system size. The adiabatic expansion ratios are shown in Figures 23 and 24, for helium and hydrogen, respectively. The total mass of a stored gas pressurization system is nearly proportional to the expansion ratio. It is therefore, necessary that this parameter be a near minimum value for an optimum weight system.

Referring to Figure 23, it is noted that there is a significant decrease in helium expansion ratio as temperature is dropped from 530°R and 140°R. Between 140°R and 37°R, however, little improvement is to be gained. In Figure 24, the hydrogen expansion ratio decreases significantly from 530°R to 140°R, then increases when the storage temperature is dropped to 70°R. These effects are caused by the deviations from the perfect gas law which both gases

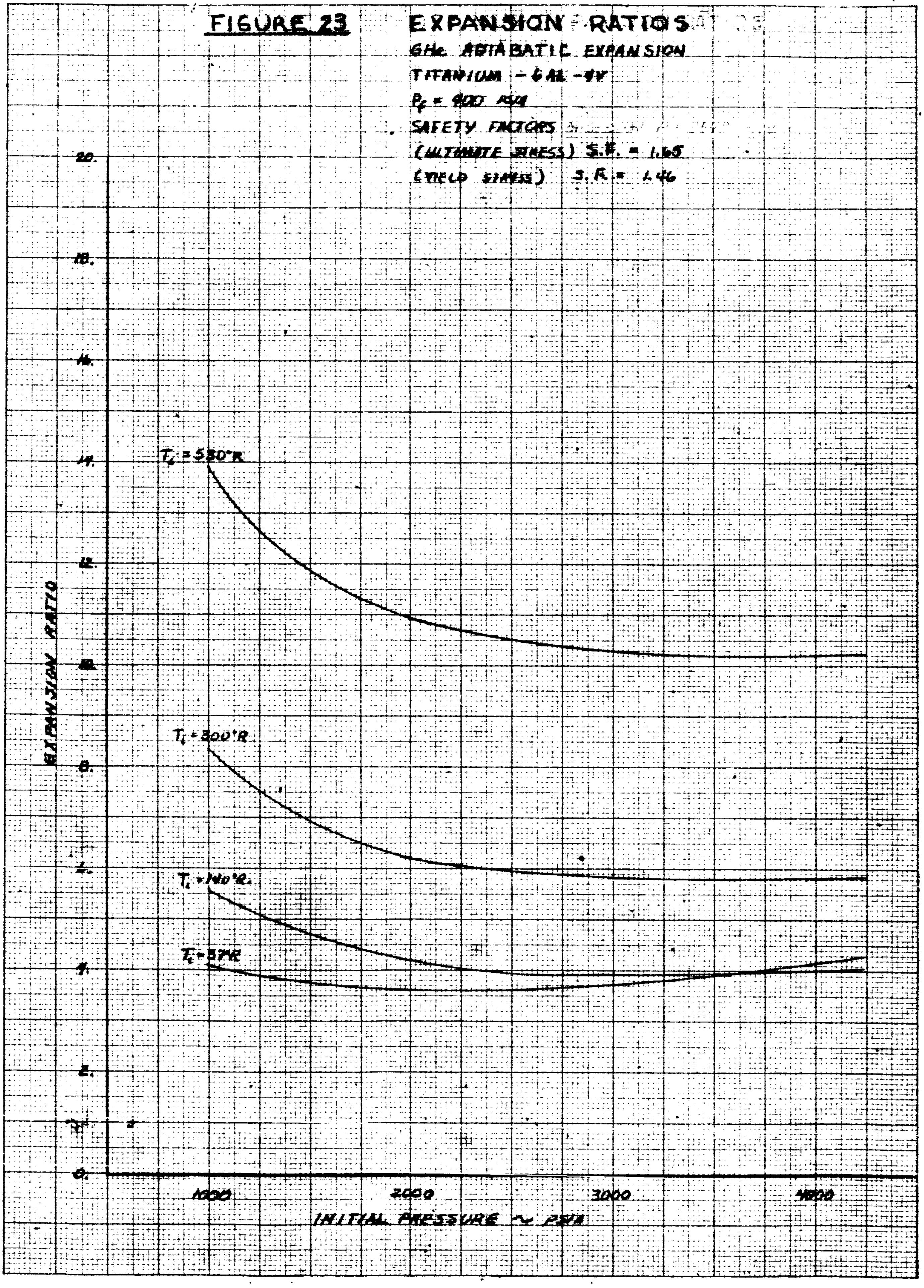


FIGURE 23
EXPANSION RATIOS
GHe ADIABATIC EXPANSION
TITANIUM - 6AL - 4V
P_f = 400 PSIA
SAFETY FACTORS
(ULTIMATE STRESS) S.F. = 1.65
(YIELD STRESS) S.F. = 1.46
20.
18.
16.
14.
12.
10.
8.
6.
4.
2.
0.
EXPANSION RATIO
Ti = 530°R
Ti = 300°R
Ti = 140°R
Ti = 37°R
1000
2000
3000
4000
INITIAL PRESSURE ~ PSIA

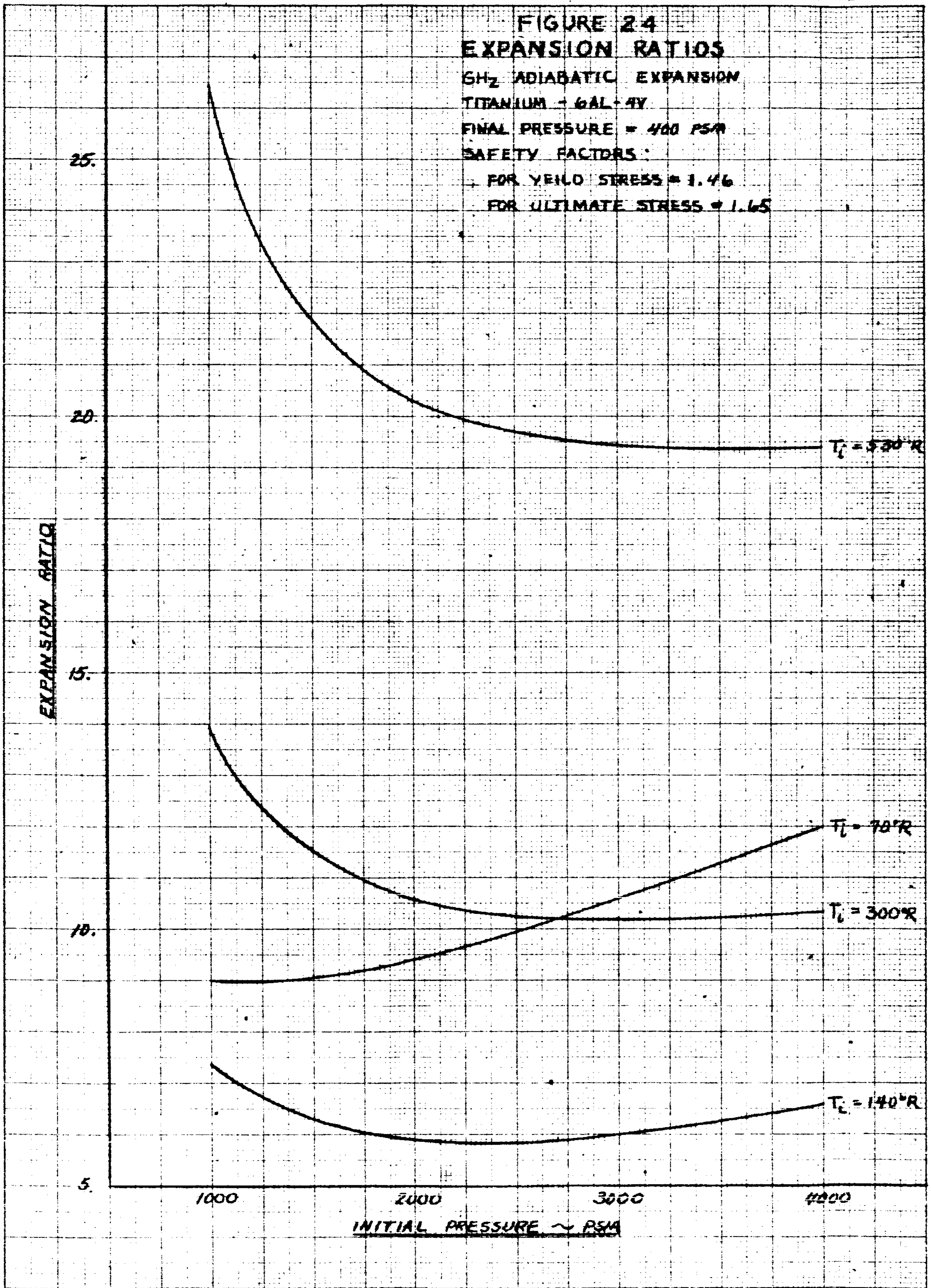


FIGURE 24
EXPANSION RATIOS
GH_2 ADIABATIC EXPANSION
TITANIUM - 6AL-4V
FINAL PRESSURE = 400 PSIA
SAFETY FACTORS:
FOR YEILD STRESS = 1.46
FOR ULTIMATE STRESS = 1.65
EXPANSION RATIO
25.
20.
15.
10.
5.
1000
2000
3000
4000
INITIAL PRESSURE ~ PSIA
T_i = 530°R
T_i = 70°R
T_i = 300°R
T_i = 140°R

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undergo as the critical conditions are approached. Figures 23 and 24 indicate that total system masses will decrease only slightly for a helium pressurization system when storage temperature is decreased from 140°R to 37°R; when hydrogen is used as the fuel pressurant, total system masses might well increase when the temperature is decreased from 140°R to 70°R.

Propellant Feed Line Heat Exchanger Analysis

The low pressure, feed line heat exchangers used in these systems are of the same basic geometry as the existing Apollo SPS heat exchangers - i.e., a single tube, counterflow coil enclosed within an expanded section of each propellant line. This analysis[2] is analogous to a single straight section of tubing which is positioned normal to the flow of propellant (corrections are being included to simulate effects of coil). The desired inlet and outlet temperatures are input into the analysis - along with the tube diameter and pertinent thermal and physical properties of propellant, gas, and tubing material. The total length of tubing required to produce the required gas temperature change is computed, as is gas pressure drop, propellant temperature change, and total mass of the heat exchanger.

The existing Apollo SPS oxidizer feed line heat exchanger design requirements were subjected to this analysis to determine validity of the model. The minimum temperature of the helium entering the heat exchanger was calculated to be 441°R. Other design Parameters

2. A more complete description of the mathematical model was transmitted to the NASA-Manned Spacecraft Center on 22 January 1965.

were:

Helium flow rate:	.07 lb_m/sec
Oxidizer flow rate:	45.3 lb_m/sec
Helium outlet pressure:	175 lb_f/in^2 absolute
Oxidizer Inlet temperature:	530°R
Tubing O.D.:	.75 inch
Tubing I.D.:	.68 inch
Tubing material:	stainless steel

The helium outlet temperature was varied in the analysis, resulting in the curves shown in Figure 25. For the actual heat exchanger mass of 8.0 lb_m and helium pressure drop of 3.5 psia, the helium outlet temperature was predicted to be about 515°R - or within 15°R of oxidizer inlet temperature. This is a favorable comparison to the existing heat exchanger, which is required to heat helium to within 25°R of propellant temperature. Additional verification of the heat exchanger analysis is discussed in Section V.

This analysis was then pursued to predict mass as a function of helium inlet temperature, for a helium outlet temperature of 515°R (within 15°R of propellant inlet temperature.) The results are shown in Figure 26. The same analysis was used to predict heat exchanger mass for the fuel feed line unit in system 1A, using hydrogen as the pressurant. These results are shown in Figure 27.

Mass Tabulations for Systems 0, 1, and 1A

The items included in the mass evaluations were heat exchangers, the storage spheres, the pressurant, valves and pressure switches. The masses of the valves and pressure switches based on existing flight hardware used on the Apollo SPS and Titan III Transtage.

FIGURE 35

OXIDIZER TO HELIUM HEAT EXCHANGER
WEIGHT AND PRESSURE DROP VS. ΔT.

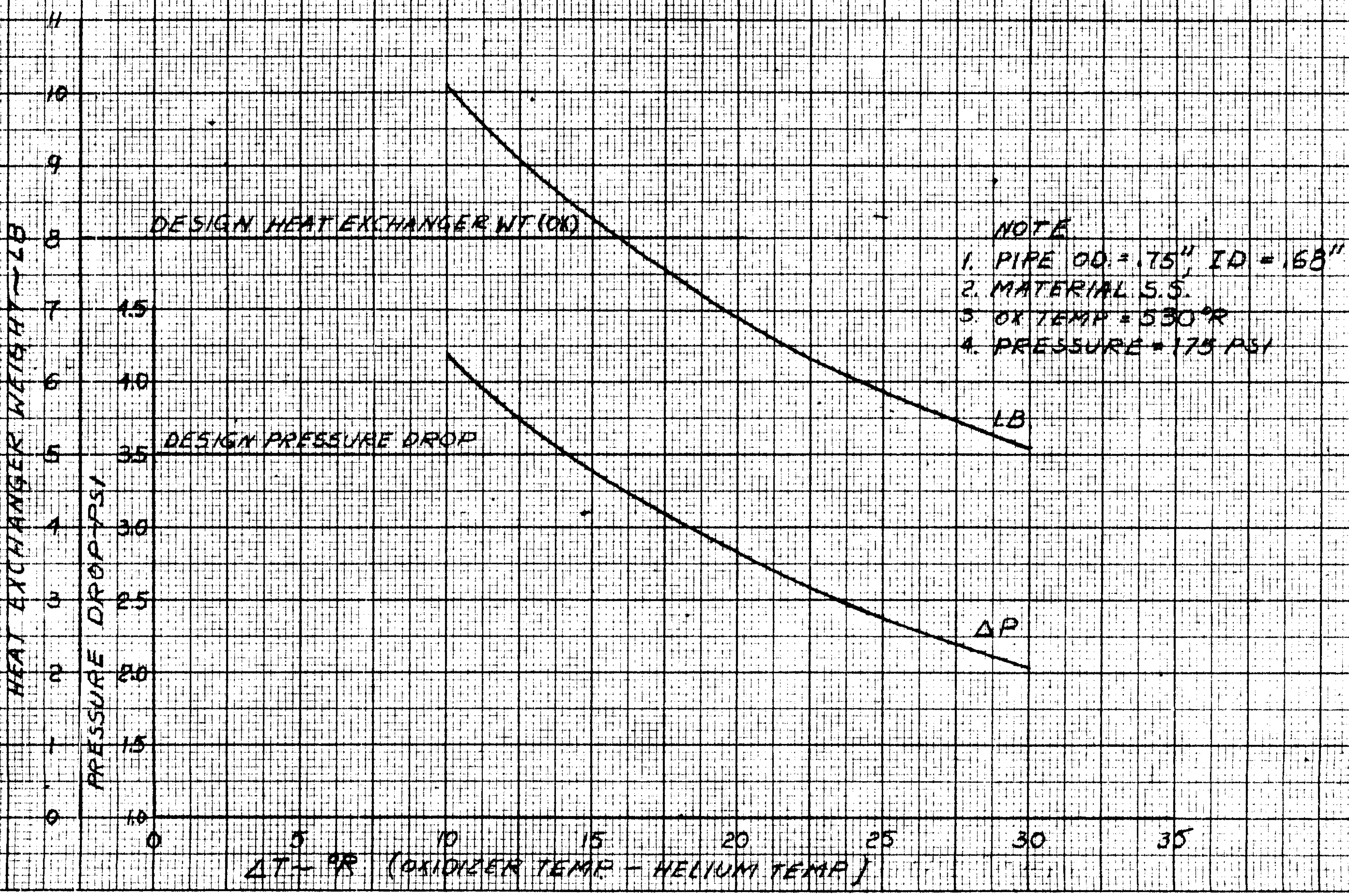


FIGURE 26

OXIDIZER OR FUEL TO HELIUM HEAT EXCHANGER
WEIGHT VS INLET TEMPERATURE

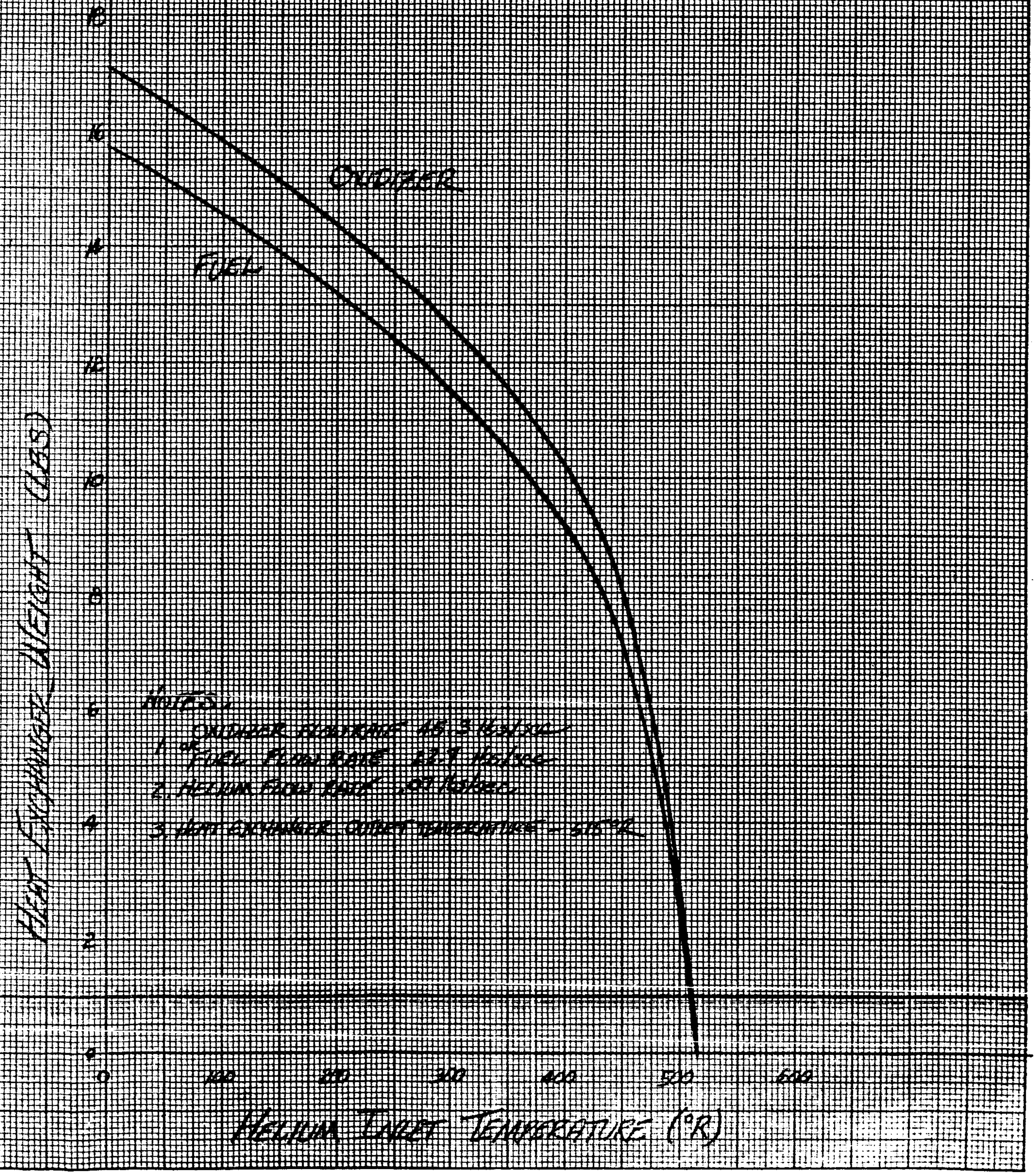
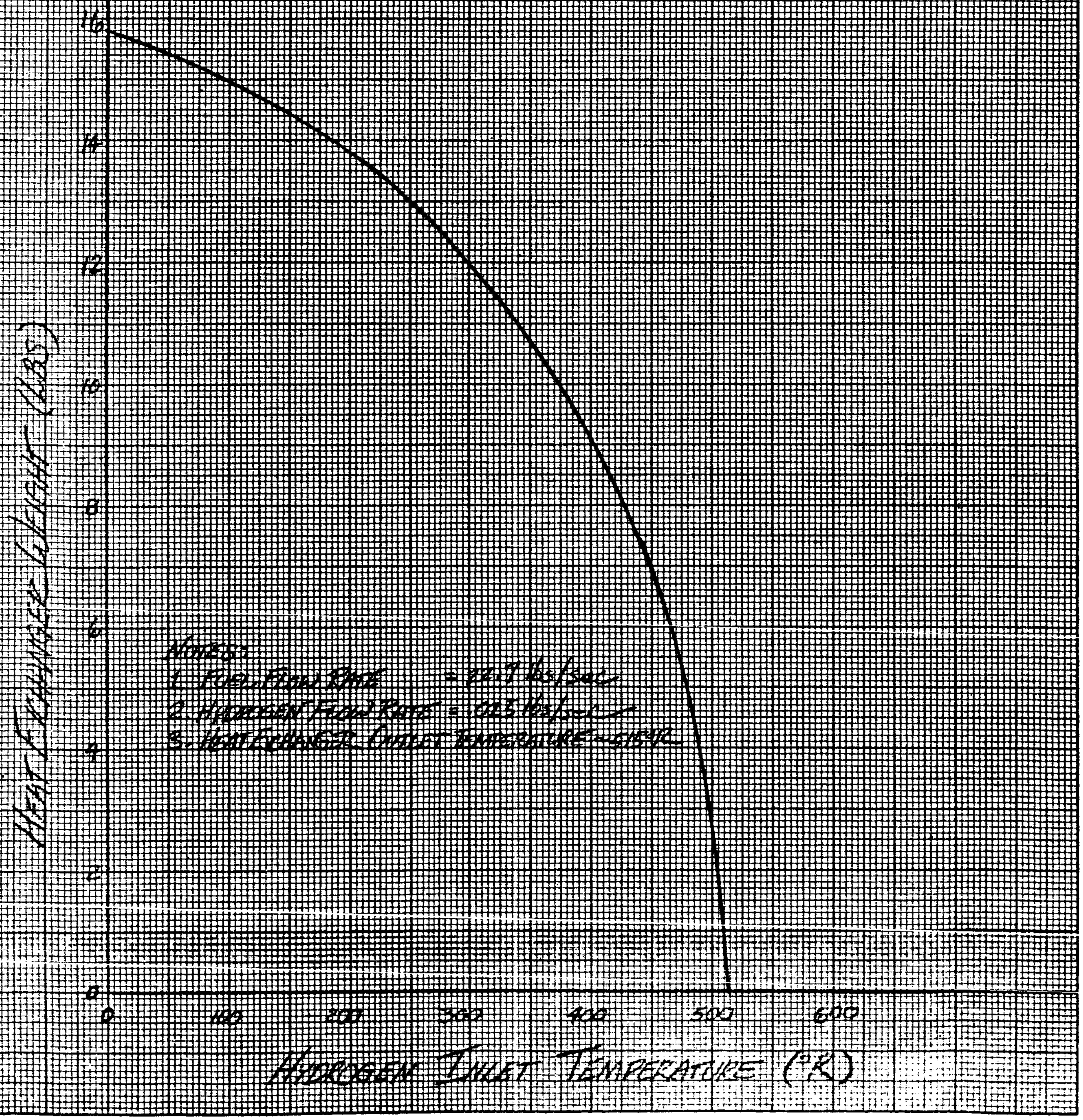


Figure 27

Fuel to Hydrogen Heat Exchanger
Weight vs Inlet Temperature



The preliminary mass estimates for systems 0, 1, and 1A are shown in Tables 3, 4, and 5, respectively; and are plotted in Figures 28 and 29. These results, along with the mass estimates for each of the other systems, are compared in Section IV.

Table 3 - System 0

TANK INLET TEMP. (°R)	HELIUM STORAGE TEMP. (°R)	HELIUM STORAGE PRESSURE (psia)	HELIUM LOADED AND STORAGE CONTAINER WT. (lbs)	TOTAL WEIGHT OF VALVES (lbs)	HEAT EXCHANGER WEIGHT (lbs)	TOTAL SYSTEM WEIGHT (lbs)
515	530	1000	892.0	28.5	15.8	936.3
515	530	2000	698.0	28.5	15.8	742.3
515	530	3000	662.0	28.5	15.8	706.3
515	530	4000	654.0	28.5	15.8	698.3

Table 4 - System 1

TANK INLET TEMP. (°R)	HELIUM STORAGE TEMP. (°R)	HELIUM STORAGE PRESSURE (psia)	HELIUM LOADED AND STORAGE CONTAINER WT. (lbs)	TOTAL WEIGHT OF VALVES (lbs)	HEAT EXCHANGER WEIGHT (lbs)	TOTAL SYSTEM WEIGHT (lbs)
515	37	1000	263.0	28.5	34.0	325.5
515	37	2000	231.0	28.5	34.0	293.5
515	37	3000	238.5	28.5	34.0	301.0
515	37	4000	266.0	28.5	34.0	328.5
520	37	1000	263.0	28.5	39.0	330.5
520	37	2000	231.0	28.5	39.0	298.5
520	37	3000	238.5	28.5	39.0	306.0
520	37	4000	266.0	28.5	39.0	333.5
515	140	1000	356.0	28.5	33.0	417.5
515	140	2000	269.0	28.5	33.0	330.5
515	140	3000	250.0	28.5	33.0	311.5
515	140	4000	256.0	28.5	33.0	317.5
515	300	1000	534.0	28.5	29.0	591.5
515	300	2000	398.0	28.5	29.0	455.5
515	300	3000	374.0	28.5	29.0	431.5
515	300	4000	372.5	28.5	29.0	430.0

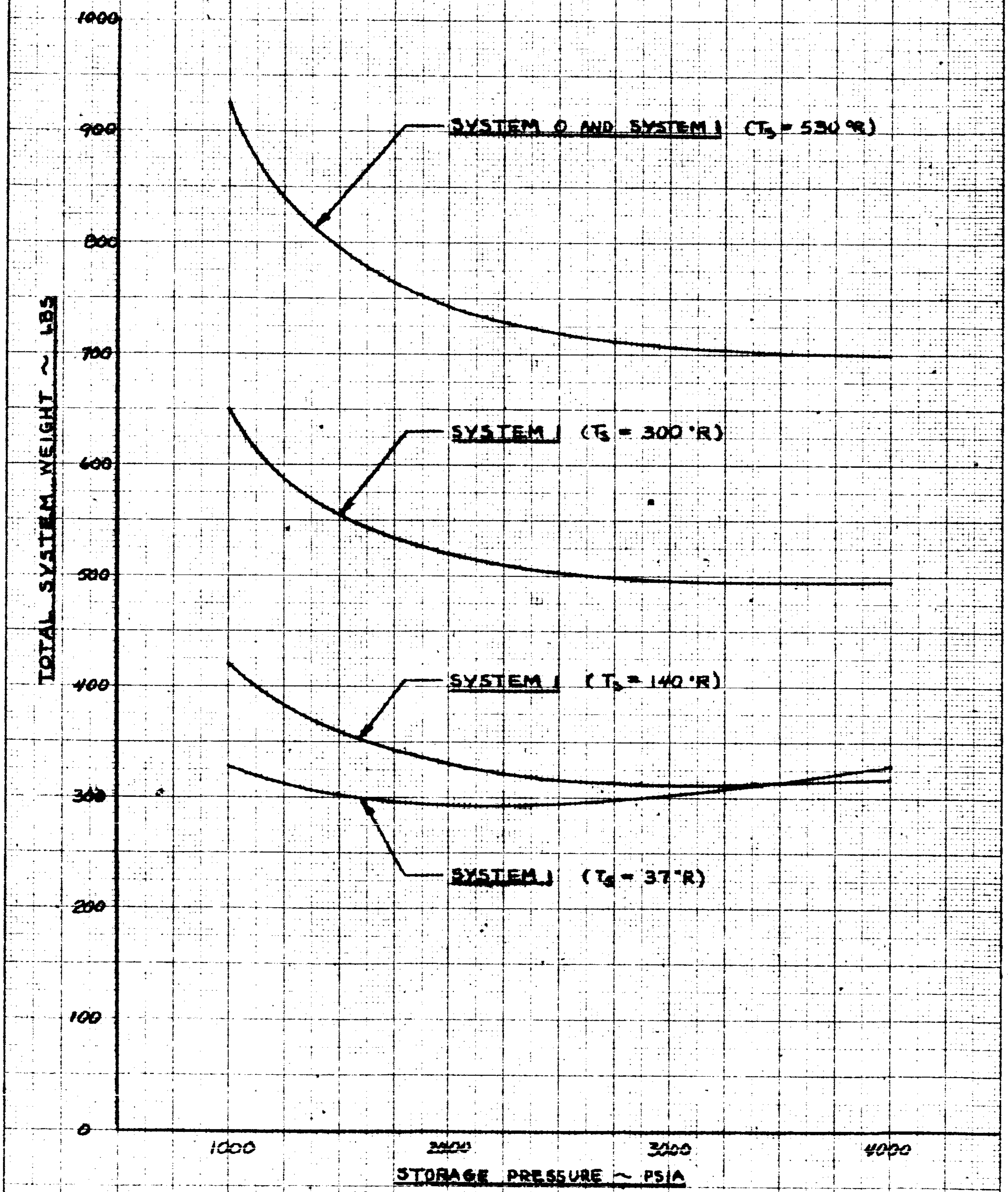
TABLE 5

SYSTEM 1 - A

TANK INLET TEMP.	PRESSURANT STORAGE TEMP.	PRESSURANT STORAGE PRESSURE	HYDROGEN LOADED AND STORAGE CONTAINER WT.	HELIUM LOADED AND STORAGE CONTAINER WT	TOTAL WEIGHT OF VALVES	HYDROGEN HEAT EX-CHANGER WT	HELIUM HEAT EX-CHANGER WT	TOTAL SYSTEM WEIGHT
515°R	70°R(GH_2) & 37°R(GHe)	1000 psia	132.926	142.773	49.0	15.5	17.0	357.199
515°R	70°R(GH_2) & 37°R(GHe)	2000 psia	139.447	125.404	49.0	15.5	17.0	346.351
515°R	70°R(GH_2) & 37°R(GHe)	3000 psia	157.378	131.657	49.0	15.5	17.0	370.535
515°R	70°R(GH_2) & 37°R(GHe)	4000 psia	178.273	144.163	49.0	15.5	17.0	403.936
515°R	140°R	1000 psia	108.579	193.109	49.0	15.5	16.5	382.688
515°R	140°R	2000 psia	86.469	144.857	49.0	15.5	16.5	312.326
515°R	140°R	3000 psia	89.803	137.979	49.0	15.5	16.5	308.782
515°R	140°R	4000 psia	97.153	138.431	49.0	15.5	16.5	316.584
515°R	300°R	1000 psia	206.429	289.923	49.0	14.0	14.5	573.852
515°R	300°R	2000 psia	157.155	214.611	49.0	14.0	14.5	449.266
515°R	300°R	3000 psia	151.984	204.607	49.0	14.0	14.5	434.091
515°R	300°R	4000 psia	153.362	202.071	49.0	14.0	14.5	432.933
515°R	530°R	1000 psia	391.073	484.005	49.0	8.0	8.0	940.080
515°R	530°R	2000 psia	300.974	378.575	49.0	8.0	8.0	744.549
515°R	530°R	3000 psia	287.933	357.280	49.0	8.0	8.0	710.213
515°R	530°R	4000 psia	287.637	354.328	49.0	8.0	8.0	706.965

FIGURE 28

TOTAL SYSTEM WEIGHTS FOR SYSTEM I



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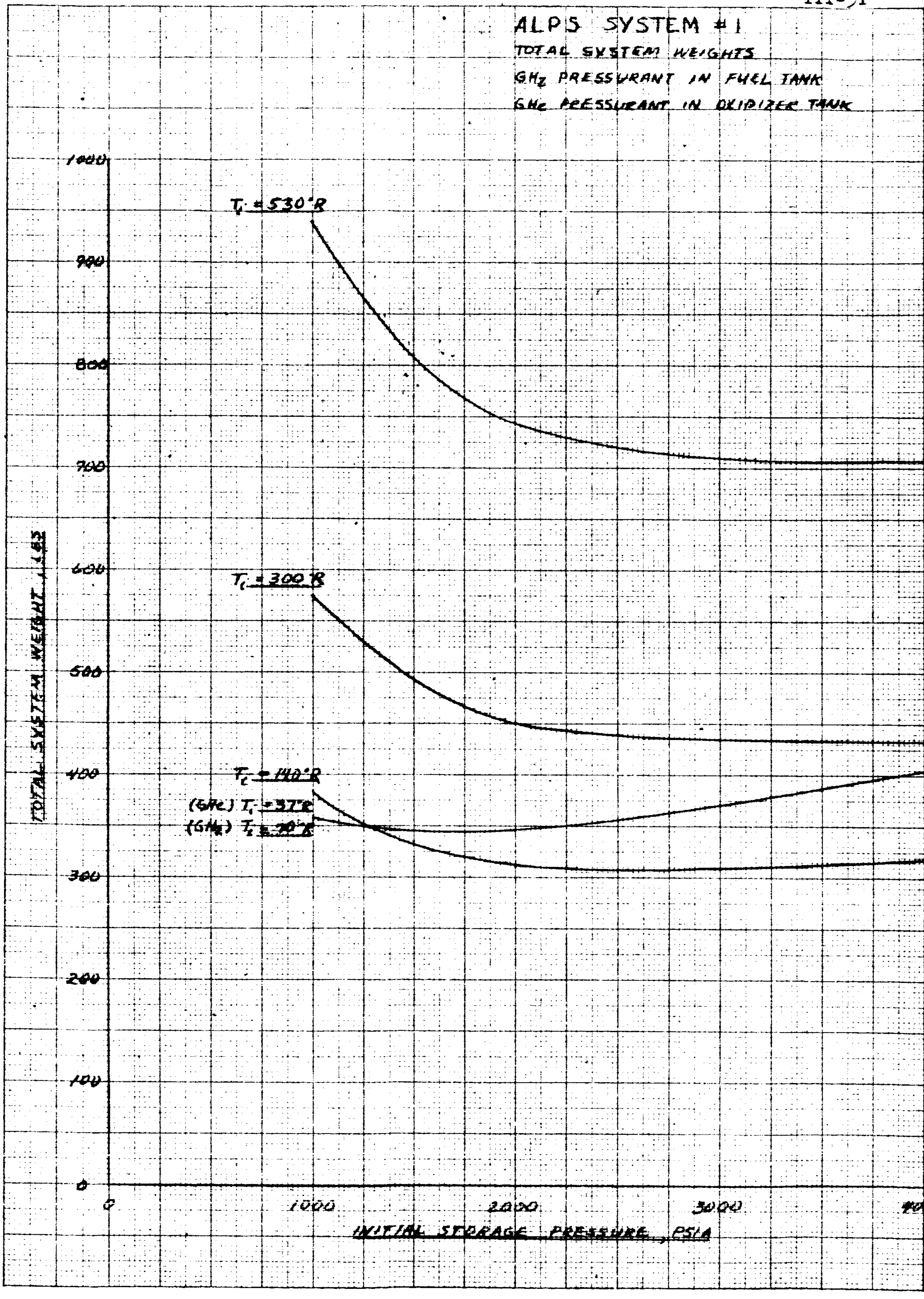

ALPS SYSTEM #1
TOTAL SYSTEM WEIGHTS
GH2 PRESSURANT IN FUEL TANK
GHe PRESSURANT IN OXIDIZER TANK
Ti = 530°R
Ti = 300°R
Ti = 140°R
(GHe) Ti = 37°R
(GH2) Ti = 30°R
TOTAL SYSTEM WEIGHT, LBS
INITIAL STORAGE PRESSURE, PSIA
1000
900
800
700
600
500
400
300
200
100
0
1000
2000
3000
4000

B. SYSTEM 2

System 2 modified the basic helium pressurant storage system by including a means of heating the helium within the storage vessel.

The purpose of this modification is to raise the final temperature of the helium and thus reduce the weight of the residual at the end of the mission. This, therefore, reduces the mass of helium loaded and the storage container mass.

For this study, a free convection heat exchanger is used, operating only during propellant burn times. It is taken to be a single straight finned tube. The heat is supplied by hot gas products of a gas generator burning main tank propellants. A constant flow rate of hot gas at a given fixed inlet temperature is assumed supplied to the heat exchanger.

The size and weight of this system are obtained from a numerical computation of the helium storage vessel thermodynamics through a mission time, as is done on the basic helium system. The major modification to existing computer programs was the inclusion of the calculation of the heat flux from the heat exchanger.

The assumptions made for the calculation include:

1) The helium storage vessel is a titanium alloy sphere, fully insulated from the environment;
2) The helium and storage vessel are at homogeneous temperatures;
3) The helium is preheated to propellant ambient temperature before entering the propellant tanks by heat exchangers in the propellant feed lines;

4) The heat capacity of the storage vessel is taken into account, and its temperature is allowed to lag behind the stored helium temperature; the heat transfer coefficient between the two is that commonly used for turbulent free convection from a vertical plate;

5) The heat exchanger is made of a high strength steel to resist collapse by the high pressure of the stored helium; the heat exchanger heat transfer coefficients are those commonly used for forced turbulent convection inside along tube and laminar free convection from a vertical plate;

6) The heat flux to the helium is calculated by numerical integration along the length of the heat exchanger; heat conduction along the heat exchanger wall is not considered here;

7) The storage vessel wall thickness and thus weight are calculated at the point of maximum helium pressure; with sufficient heat flux into the storage vessel this point occurred at a point well within the mission time;

8) During coast periods the temperatures of the stored helium, storage vessel and heat exchanger equilibrate; for this study the heat capacity of the heat exchanger is not reduced in the calculation of the coast period equilibrium temperature;

9) The thermophysical properties of the helium, storage vessel, heat exchanger and hot gas generator products are claculated by empirical relations with the exception of the helium specific heats which are inputed as constants;

10) The minimum helium storage pressure allowed is 400 psia.

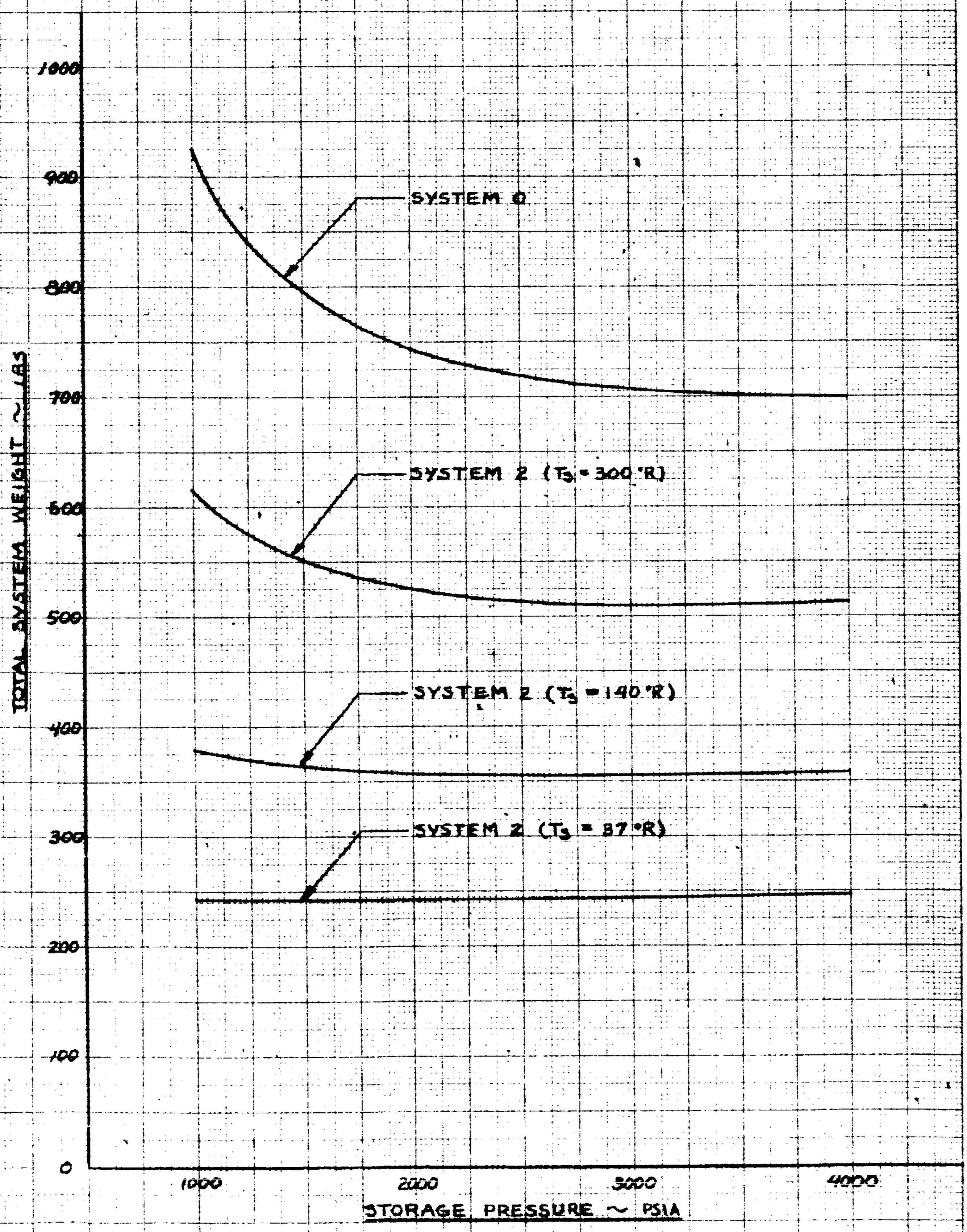
A limited optimization has been made on the system weight. Initial helium storage pressure and temperature have been varied from 1000 psia to 4000 psia and 37°R to 300°R. The heat exchanger design parameters: tube length, tube diameter and hot gas flow rate have also been varied. The results of this analysis are shown in Table 6 and Figure 30.

Table 6 - SYSTEM 2

TANK INLET TEMP.	HELIUM STORAGE TEMP.	HELIUM STORAGE PRESSURE	HELIUM LOADED AND STORAGE CONTAINER WT	HELIUM CONTAINER COIL WEIGHT	GAS GENERATOR WEIGHT	GAS GENERATOR REACTANTS WEIGHT	TOTAL WEIGHT OF VALVES	HEAT EX-CHANGER WEIGHT	TOTAL SYSTEM WEIGHT
515°R	37°R	1000 psia	115.2 lbs	30.8 lbs	8.0 lbs	17.7 lbs	37.5 lbs	34.0 lbs	243.2 lbs
515°R	37°R	2000 psia	114.4 lbs	30.8 lbs	8.0 lbs	17.7 lbs	37.5 lbs	34.0 lbs	242.4 lbs
515°R	37°R	3000 psia	116.1 lbs	30.8 lbs	8.0 lbs	17.7 lbs	37.5 lbs	34.0 lbs	244.1 lbs
515°R	37°R	4000 psia	119.2 lbs	30.8 lbs	8.0 lbs	17.7 lbs	37.5 lbs	34.0 lbs	247.2 lbs
515°R	140°R	1000 psia	211.4 lbs	30.8 lbs	8.0 lbs	59.0 lbs	37.5 lbs	33.0 lbs	379.7 lbs
515°R	140°R	2000 psia	189.0 lbs	30.8 lbs	8.0 lbs	59.0 lbs	37.5 lbs	33.0 lbs	357.3 lbs
515°R	140°R	3000 psia	188.6 lbs	30.8 lbs	8.0 lbs	59.0 lbs	37.5 lbs	33.0 lbs	356.9 lbs
515°R	140°R	4000 psia	189.8 lbs	30.8 lbs	8.0 lbs	59.0 lbs	37.5 lbs	33.0 lbs	358.1 lbs
515°R	300°R	1000 psia	450.9 lbs	30.8 lbs	8.0 lbs	59.0 lbs	37.5 lbs	29.0 lbs	615.2 lbs
515°R	300°R	2000 psia	357.4 lbs	30.8 lbs	8.0 lbs	59.0 lbs	37.5 lbs	29.0 lbs	521.7 lbs
515°R	300°R	3000 psia	345.2 lbs	30.8 lbs	8.0 lbs	59.0 lbs	37.5 lbs	29.0 lbs	509.5 lbs
515°R	300°R	4000 psia	346.3 lbs	30.8 lbs	8.0 lbs	59.0 lbs	37.5 lbs	29.0 lbs	510.6 lbs

FIGURE 30

TOTAL SYSTEM WEIGHTS FOR SYSTEM 2



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C. SYSTEM 4

System 4 is similar to system 2; the basic difference being the use of a high temperature gas-to-gas heat exchanger in place of the ambient temperature propellant feed line heat exchangers. The heat source is a bipropellant (N_2O_4/.5 N_2H_4-.5 UDMH) gas generator which are supplied by propellants from the main SPS tanks. The propellant tank helium usage and helium tank heat exchanger analyses were performed as described in Sections III-A and III-B, respectively. The only other significant analytical effort involved in system 4 is the heat exchanger/gas generator analysis, which is discussed below.

Heat Exchanger/Gas Generator Analysis

An existing mathematical model, programmed for use with the IBM 1620 computer, was used to predict heat exchanger and gas generator weight, for various conditions of entrance and exit temperatures, pressures, and gas flow rates. The model is based upon certain methods presented in the book Compact Heat Exchangers, by W. M. Kays and A. L. London. The basic configuration chosen for this analysis is a cross flow, finned tube unit with the fins exposed to the hot gas and the cold helium flowing inside the smooth tubing. Figure 92 of Kays and London is typical of this type of heat exchanger, and the functional relationships presented in that figure were used to define the hot gas film coefficients. Internal film coefficients for the cold gas were based upon the turbulent portion of Figure 41 in Kays and London.

The gas generator/heat exchanger analysis was approached from two directions.

1) Constant initial pressurant temperature at fixed pressurant flow rates. This method provided consistent curves of: (heat exchanger mass + gas generator mass + total hot gas mass) versus temperature rise of the pressurant. These results are shown in Table 7 and Figure 31.

2) Actual system mass requirements for specific exit temperatures, inlet temperatures and initial storage pressures. Storage pressures 1000, 2000, 3000, and 4000 psia were evaluated in terms of the reduced heat exchanger inlet temperatures resulting from gas expansion in insulated storage vessels. Pressurant gas flow rate requirements which varied with the heat exchanger outlet temperatures were considered in this analysis. It was considered that this approach to the analysis was more realistic than the above, and was used in the final mass analysis. These results are shown in Table 8 and Figure 32.

The hot gas combustion products were based upon operation of the gas generator at an oxidizer to fuel mass ratio of 0.103. This ratio produces gas at a temperature within materials limitations, and results in stable operation of the gas generator. The pertinent properties of the combustion products (specific heat, thermal conductivity, viscosity, etc.) were input to the analysis as functions of temperature.

TABLE 7

GAS TO GAS HEAT EXCHANGER DATA

INPUT VALUES							CALCULATED VALUES				
RUN NO.	PRESS. PSIA	INITIAL TEMP. °R	FINAL TEMP.	W_{He} #/SEC	XNTU	ΔT	HEAT EXCH. WT. LB.	GAS GEN. WT. LB.	W_{PROP} #/SEC	WT OF PROP. LB	TOTAL WEIGHT
1	400	37	300	.15846	1.853	263	6.851	8.0	.053	47.2	62.051
2			400		1.635	363	7.683	8.0	.079	70.3	85.983
3			500		1.450	463	7.956	8.0	.108	96.2	112.056
4			600		1.32	563	8.598	8.0	.142	126.3	142.898
5		37	750		1.111	713	8.742	8.0	.204	181.7	198.442
6		100	300		1.995	200	6.135	8.0	.040	35.6	49.735
7			400		1.732	300	7.170	8.0	.065	57.9	72.070
8			500		1.528	400	7.522	8.0	.093	82.8	98.322
9			600		1.358	500	8.103	8.0	.126	112.2	128.303
10		100	750		1.15	650	8.466	8.0	.186	165.5	181.966
11		140	300		2.11	160	5.613	7.9	.032	28.5	42.013
12			400		1.808	260	6.815	8.0	.056	49.8	64.615
13			500		1.58	360	7.250	8.0	.084	74.8	90.050
14			600		1.41	460	7.985	8.0	.116	103.3	119.285
15		140	750		1.18	610	8.265	8.0	.175	156.0	172.265
16		300	350		2.56	50	3.244	7.9	.010	8.90	20.044
17			400		2.27	100	4.592	7.9	.021	18.7	31.192
18			500		1.887	200	5.885	8.0	.046	40.9	54.785
19			600		1.625	300	6.900	8.0	.075	66.7	82.6
20	400	300	750	.15846	1.315	450	7.434	8.0	.129	114.9	130.334

FIGURE 31

TOTAL WEIGHT OF HEAT EXCHANGER, GAS GENERATOR, PROPELLANT VS TEMPERATURE DIFFERENCE ACROSS HEAT EXCHANGER

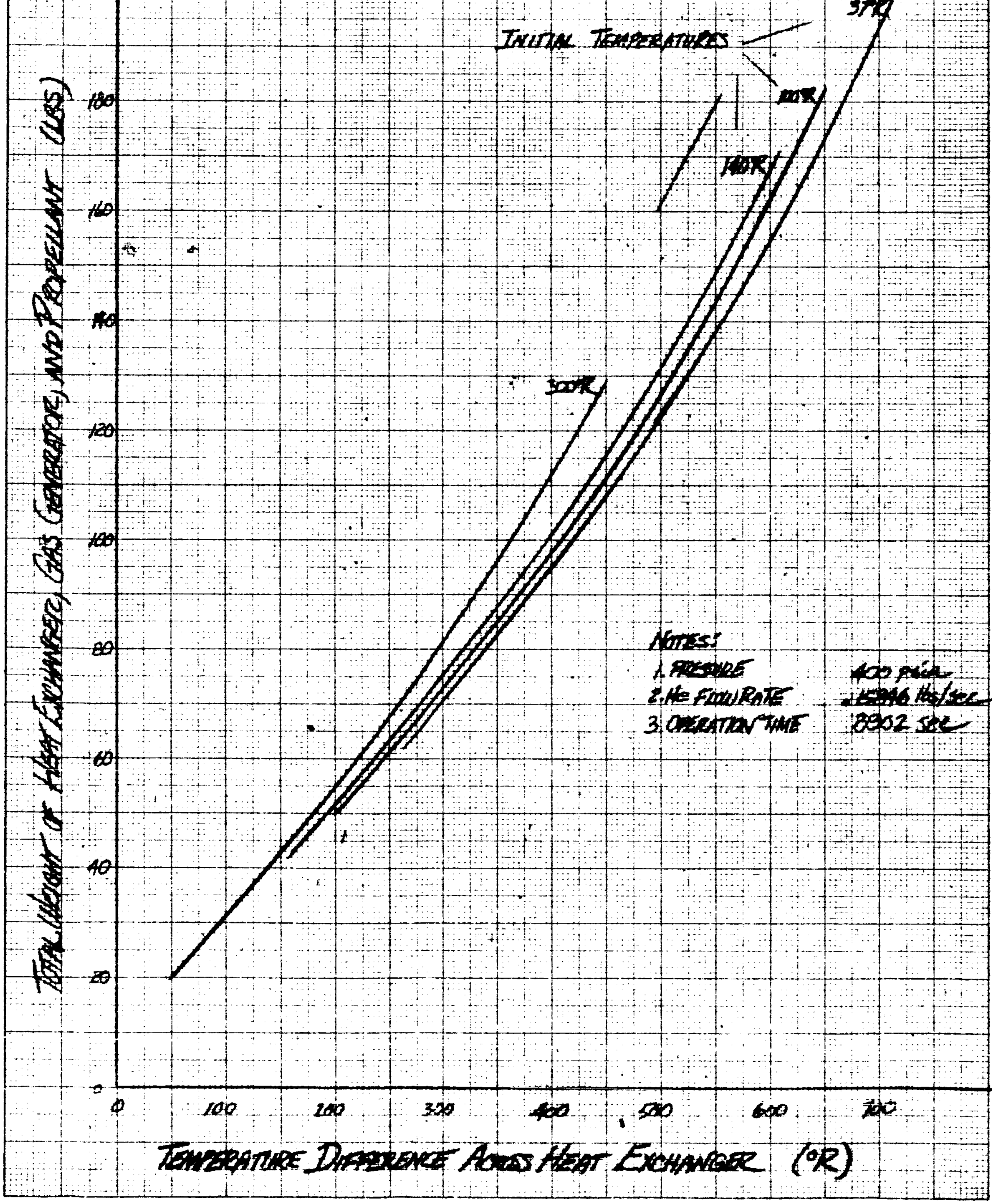
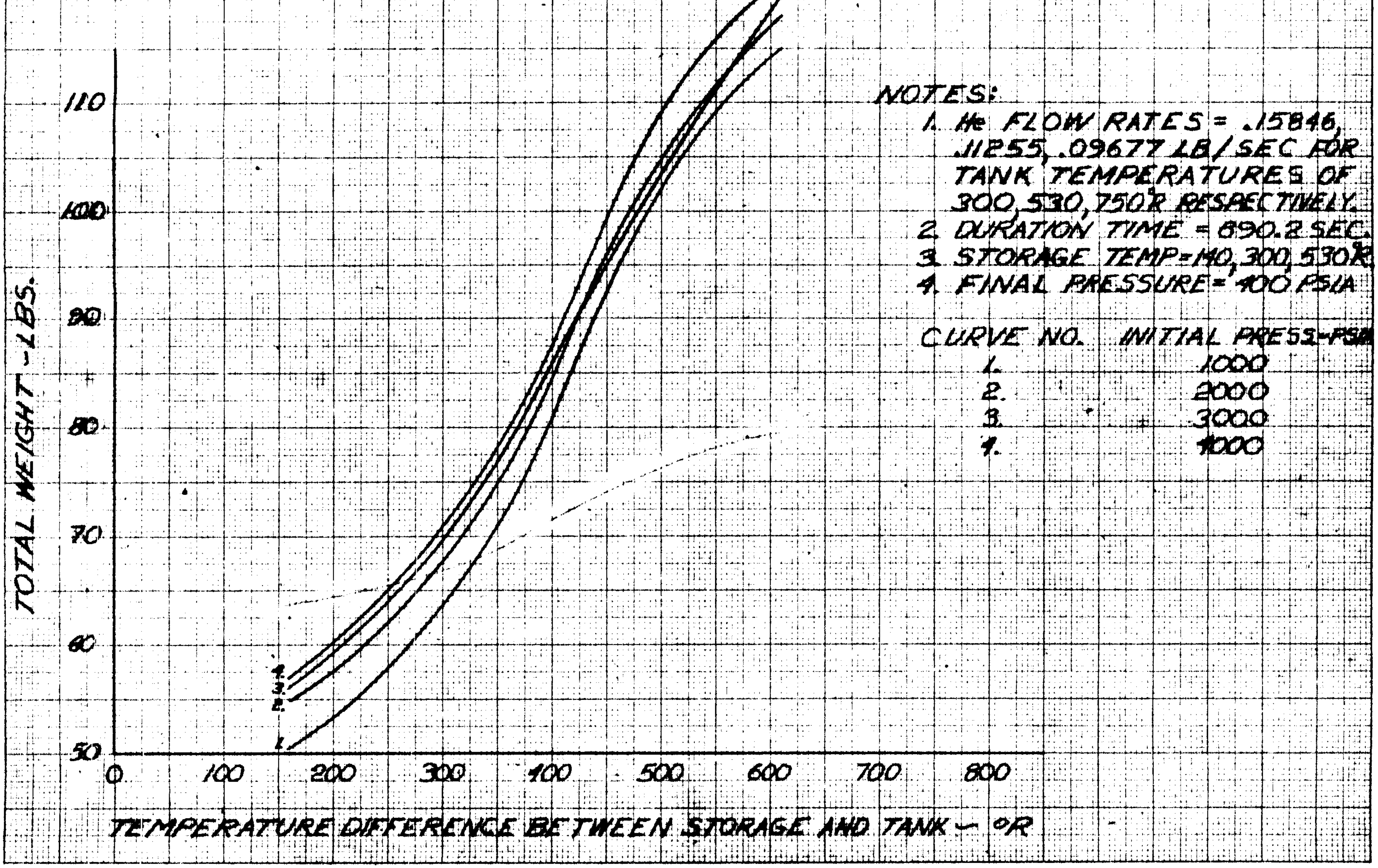


TABLE 8

GAS TO GAS HEAT EXCHANGER DATA

INPUT VALUES						CALCULATED VALUES							
RUN NO.	PRESS. DROP PSIA	HELIUM STORAGE TEMP. °R	TEMP. IN PROPELLANT TANKS °R	MINIMUM TEMPERATURE °R	$\dot{W}_{He}$ MAX #/SEC	NTU	ΔT	HEAT EXCH. WEIGHT (LB)	GAS GEN. WT. (LB)	$\dot{W}_{prop}$ #/SEC	WT OF PROP (LB)	TOTAL WEIGHT
1	1000 —400	140	300	100	.15846	2.00	160	6.158	8.0	.041	36.5	50.658
2	2000 —400			78.1		1.95		6.455	8.0	.045	40.1	54.555
3	3000 —400			67.9		1.92		6.538	8.0	.047	41.8	56.338
4	4000 —400	140	300	61.7	.15846	1.90	160	6.574	8.0	.048	42.7	57.274
5	1000 —400	530	750	486.6	.09674	1.60	220	4.580	8.0	.046	40.9	53.48
6	2000 —400			442.5		1.52		4.817	8.0	.053	47.2	60.017
7	3000 —400			430.5		1.50		4.872	8.0	.055	48.9	61.77
8	4000 —400	530	750	424.0	.09674	1.49	220	4.904	8.0	.057	50.7	63.604
9	1000 —400	300	530	247.4	.11255	1.69	230	5.158	8.0	.048	42.7	55.858
10	2000 —400			220.8		1.652		5.721	8.0	.052	46.3	60.021
11	3000 —400			213.3		1.625		5.511	8.0	.053	47.2	60.711
12	4000 —400	300	530	202.8	.11255	1.620	230	5.630	8.0	.055	49.0	62.630
13	1000 —400	140	530	100	.11255	1.472	390	6.111	8.0	.073	65.0	79.111
14	2000 —400			78.1		1.440		6.165	8.0	.076	67.6	81.765
15	3000 —400			67.9		1.435		6.325	8.0	.078	69.4	83.725
16	4000 —400	140	530	61.7	.11255	1.433	390	6.383	8.0	.079	70.3	84.683
17	1000 —400	300	750	247.4	.09677	1.278	450	5.630	8.0	.088	78.4	92.03
18	2000 —400			220.8		1.245		5.659	8.0	.092	81.9	95.559
19	3000 —400			213.3		1.241		5.700	8.0	.094	83.6	97.3
20	4000 —400	300	750	202.8	.09677	1.235	450	5.755	8.0	.095	84.5	98.255
21	1000 —400	140	750	100	.09677	1.150	610	6.009	8.0	.113	100.8	114.809
22	2000 —400			78.1		1.135		6.067	8.0	.117	104.1	118.167
23	3000 —400			67.9		1.129		6.100	8.0	.119	106.0	120.1
24	4000 —400	140	750	61.7	.09677	1.128	610	6.116	8.0	.12	106.9	121.016

FIGURE 32
TOTAL WEIGHT OF HEAT EXCHANGER, GAS GENERATOR AND PROPELLANT
VS
TEMPERATURE DIFFERENCE BETWEEN STORAGE AND TANK



Tables 7 and 8 show that the sum of gas generator mass and heat exchanger mass is nearly constant for any condition. The major variation occurs in the gas generator propellant mass. Figures 31 and 32 illustrate that total (gas generator + heat exchanger + propellant) mass increases directly with temperature rise of the pressurant, although not linearly. Nonlinearities in the data result from variations in gas properties with temperature, effects of expansion cooling in the storage sphere, and variations in gas flow rates with temperature.

Total system masses for system 4 are shown in Table 9 and Figure 33.

TABLE 9 - SYSTEM 4

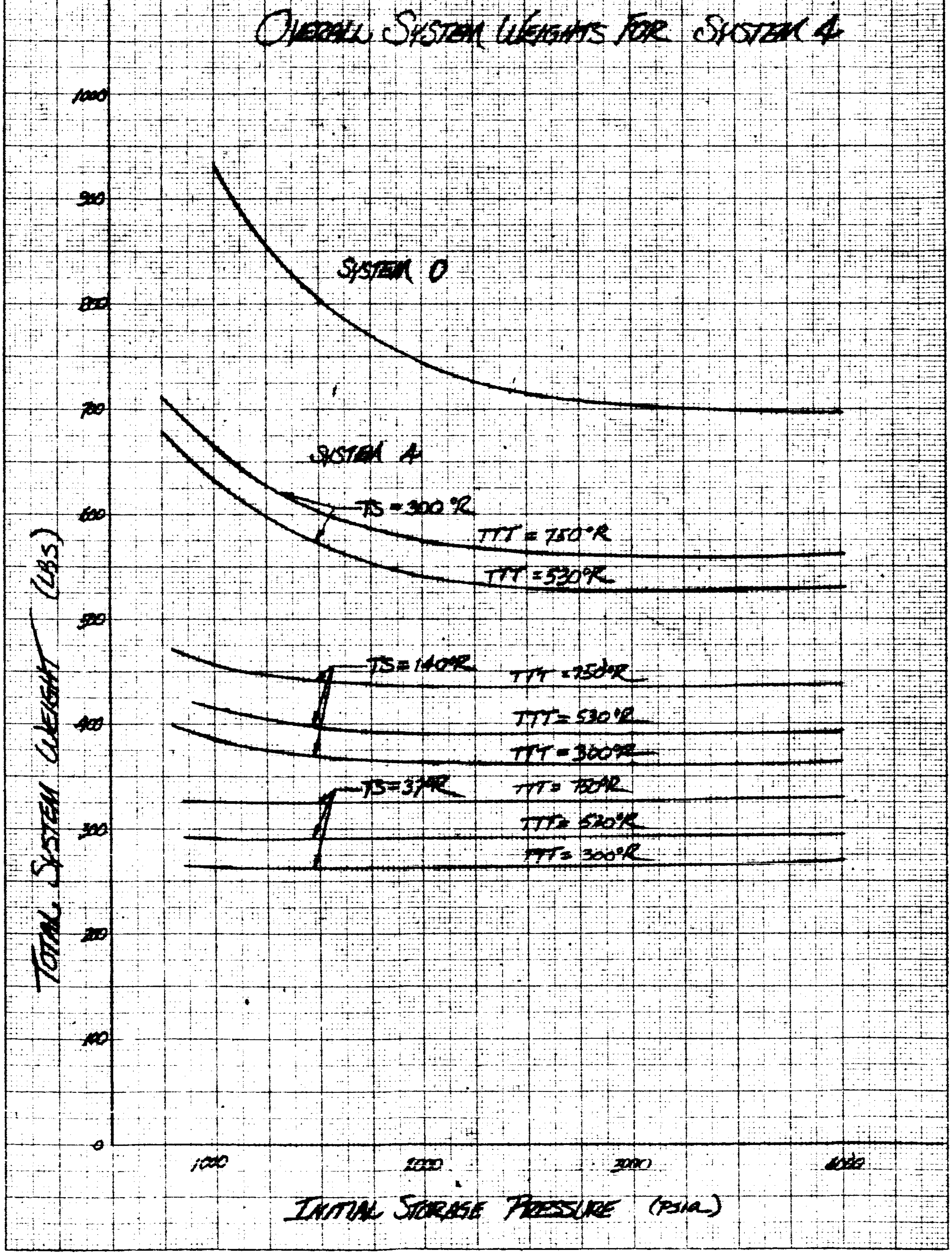
TANK INLET TEMP.	HELIUM STORAGE TEMP.	HELIUM STORAGE PRESSURE	HELIUM LOADED AND STORAGE CONTAINER WT.	HELIUM CONTAINER COIL WT.	GAS GENERATOR WT.	GAS GENERATOR REACTANTS WEIGHT	TOTAL WEIGHT OF VALVES	HEAT EXCHANGER WEIGHT	TOTAL SYSTEM WEIGHT
750°R	37°R	1000 psia	113.7 lbs	30.8 lbs	16.0 lbs	118.3 lbs	37.5 lbs	8.74 lbs	325.04 lbs
750°R	37°R	2000 psia	112.9 lbs	30.8 lbs	16.0 lbs	118.3 lbs	37.5 lbs	8.80 lbs	324.30 lbs
750°R	37°R	3000 psia	114.6 lbs	30.8 lbs	16.0 lbs	118.3 lbs	37.5 lbs	9.00 lbs	326.20 lbs
750°R	37°R	4000 psia	117.7 lbs	30.8 lbs	16.0 lbs	118.3 lbs	37.5 lbs	9.10 lbs	329.40 lbs
530°R	37°R	1000 psia	114.9 lbs	30.8 lbs	16.0 lbs	82.0 lbs	37.5 lbs	8.07 lbs	289.27 lbs
530°R	37°R	2000 psia	114.1 lbs	30.8 lbs	16.0 lbs	82.0 lbs	37.5 lbs	8.17 lbs	288.57 lbs
530°R	37°R	3000 psia	115.8 lbs	30.8 lbs	16.0 lbs	82.0 lbs	37.5 lbs	8.27 lbs	290.37 lbs
530°R	37°R	4000 psia	118.9 lbs	30.8 lbs	16.0 lbs	82.0 lbs	37.5 lbs	8.37 lbs	293.57 lbs
300°R	37°R	1000 psia	118.3 lbs	30.8 lbs	16.0 lbs	52.7 lbs	37.5 lbs	6.85 lbs	263.15 lbs
300°R	37°R	2000 psia	117.5 lbs	30.8 lbs	16.0 lbs	52.7 lbs	37.5 lbs	6.95 lbs	262.45 lbs
300°R	37°R	3000 psia	119.2 lbs	30.8 lbs	16.0 lbs	52.7 lbs	37.5 lbs	7.05 lbs	264.25 lbs
300°R	37°R	4000 psia	122.3 lbs	30.8 lbs	16.0 lbs	52.7 lbs	37.5 lbs	7.15 lbs	267.45 lbs
750°R	140°R	1000 psia	209.9 lbs	30.8 lbs	16.0 lbs	157.0 lbs	37.5 lbs	6.01 lbs	457.21 lbs
750°R	140°R	2000 psia	187.5 lbs	30.8 lbs	16.0 lbs	157.0 lbs	37.5 lbs	6.07 lbs	434.87 lbs
750°R	140°R	3000 psia	187.1 lbs	30.8 lbs	16.0 lbs	157.0 lbs	37.5 lbs	6.10 lbs	434.50 lbs
750°R	140°R	4000 psia	188.3 lbs	30.8 lbs	16.0 lbs	157.0 lbs	37.5 lbs	6.12 lbs	435.72 lbs

TABLE 9 (continued)

SYSTEM 4

TANK INLET TEMP.	HELIUM STORAGE TEMP.	HELIUM STORAGE PRESSURE	HELIUM LOADED AND STORAGE CONTAINER WT.	HELIUM CONTAINER COIL WT.	GAS GENERATOR WEIGHT	GAS GENERATOR REACTANTS WEIGHTS	TOTAL WEIGHT OF VALVES	HEAT EXCHANGER WEIGHT	TOTAL SYSTEM WEIGHT
530°R	140°R	1000 psia	191.1 lbs	30.8 lbs	16.0 lbs	112.0 lbs	37.5 lbs	6.11 lbs	413.51 lbs
530°R	140°R	2000 psia	188.7 lbs	30.8 lbs	16.0 lbs	112.0 lbs	37.5 lbs	6.20 lbs	391.20 lbs
530°R	140°R	3000 psia	188.3 lbs	30.8 lbs	16.0 lbs	112.0 lbs	37.5 lbs	6.30 lbs	390.90 lbs
530°R	140°R	4000 psia	189.5 lbs	30.8 lbs	16.0 lbs	112.0 lbs	37.5 lbs	6.40 lbs	392.20 lbs
300°R	140°R	1000 psia	214.5 lbs	30.8 lbs	16.0 lbs	80.2 lbs	37.5 lbs	6.16 lbs	385.16 lbs
300°R	140°R	2000 psia	192.1 lbs	30.8 lbs	16.0 lbs	80.2 lbs	37.5 lbs	6.46 lbs	363.06 lbs
300°R	140°R	3000 psia	191.7 lbs	30.8 lbs	16.0 lbs	80.2 lbs	37.5 lbs	6.54 lbs	362.74 lbs
300°R	140°R	4000 psia	192.9 lbs	30.8 lbs	16.0 lbs	80.2 lbs	37.5 lbs	6.57 lbs	363.97 lbs
750°R	300°R	1000 psia	449.4 lbs	30.8 lbs	16.0 lbs	127.8 lbs	37.5 lbs	5.63 lbs	667.13 lbs
750°R	300°R	2000 psia	355.9 lbs	30.8 lbs	16.0 lbs	127.8 lbs	37.5 lbs	5.66 lbs	573.66 lbs
750°R	300°R	3000 psia	343.7 lbs	30.8 lbs	16.0 lbs	127.8 lbs	37.5 lbs	5.70 lbs	561.50 lbs
750°R	300°R	4000 psia	344.8 lbs	30.8 lbs	16.0 lbs	127.8 lbs	37.5 lbs	5.76 lbs	562.66 lbs
530°R	300°R	1000 psia	450.6 lbs	30.8 lbs	16.0 lbs	92.05 lbs	37.5 lbs	5.16 lbs	632.11 lbs
530°R	300°R	2000 psia	357.1 lbs	30.8 lbs	16.0 lbs	92.05 lbs	37.5 lbs	5.72 lbs	539.17 lbs
530°R	300°R	3000 psia	344.9 lbs	30.8 lbs	16.0 lbs	92.05 lbs	37.5 lbs	5.51 lbs	526.76 lbs
530°R	300°R	4000 psia	346.0 lbs	30.8 lbs	16.0 lbs	92.05 lbs	37.5 lbs	5.63 lbs	527.99 lbs

FIGURE 33


OVERALL SYSTEM WEIGHTS FOR SYSTEM 4
SYSTEM 0
SYSTEM 4
TS = 300 °R
TTT = 750 °R
TTT = 530 °R
TS = 140 °R
TTT = 750 °R
TTT = 530 °R
TTT = 300 °R
TS = 37 °R
TTT = 750 °R
TTT = 530 °R
TTT = 300 °R
TOTAL SYSTEM WEIGHT (LBS)
1000
900
800
700
600
500
400
300
200
100
0
1000
2000
3000
4000
INITIAL STORAGE PRESSURE (PSIA)

D. SYSTEM 5

System 5 represents the "cascade" pressurization technique wherein cold helium bled from the primary storage tank is replaced with warmer helium from a cascaded storage tank. The weight analysis and calculations that were followed in determining the total system weights for system 5 are explained below. Also, the mathematical model used to simulate that cascade expansion processes is briedly explained. A more complete description of this program was submitted to NASA-Manned Spacecraft Center in May, 1965 (Martin CR-65-36, Preliminary Utilization Instructions - Cascade Pressurization Computer Program). Initial primary storage container temperatures investigated were 300, 140, and 37°R. The tank inlet temperature was chosen at 530°R which established the mass of helium expelled from the primary storage container at 64.2 pounds. Initial primary storage container pressures considered were 1000, 2000, 3000, and 4000 psia. The initial storage temperature for the cascade container was set at 530°R.

The computer program simulated the helium expansion process out of the primary container. The program, also, simulated the expansion process from the cascade container into the primary container simultaneously with the helium expansion process out of the primary container. The program first calculated the primary storage container mass based upon the particular material properties, safety factors, and required volume. To simulate the expansion process, the program expanded a required amount of cold helium for each pressurization event. This single expansion continues until the primary storage container pressure drops to 300 psia. At that pressure and in order to maintain that pressure, ambient helium was expanded from the cascade container into the primary container simultaneously with the cold gas expansion out of the primary container. Heat transfer between the primary container wall and both the hot and cold helium was considered. Also, the heat transfer between the hot helium and cold helium was considered. After each pressurization event, the primary container wall, hot helium, and cold helium were forced into thermal equilibrium during each coast period. When all the cold helium was expelled, the mission was completed and only the warmer cascaded helium was left in the primary container. From the mass of cascaded helium left, the cascade container mass and the initial mass of helium in the cascade container can be determined. In order to obtain the cascade container mass and helium load mass, three requirements were that the cascade expansion process be adiabatic, the initial storage container temperature be 530°R, and the final cascade container pressure be 400 psia. Because of these requirements, an optimum expansion ratio was obtained from the adiabatic expansion ratio curves for helium, Figure 23 of this report. For an initial storage temperature of 530°R, the minimum expansion ratio is 10.2 at an initial storage pressure of 4000 psia. The expansion ratio of 10.2 was multiplied by the mass of helium expelled to obtain the sum of the initial helium loaded for the cascade container and the mass of the cascade storage container.

A mathematical model, discussed in Section III-A, was used to simulate a liquid-to-gas heat exchanger. This program was used to calculate the feed line heat exchanger weights for this system.

The total system weight for the cascade pressurization system, system 5, was calculated by adding the weights of the following components:

1) primary storage container,
2) primary pressurant expelled,
3) cascade storage container and cascade pressurant loaded,
4) feed line heat exchangers,
5) valves.

The total system weights for system 5 are shown in Table 10 and Figure 34.

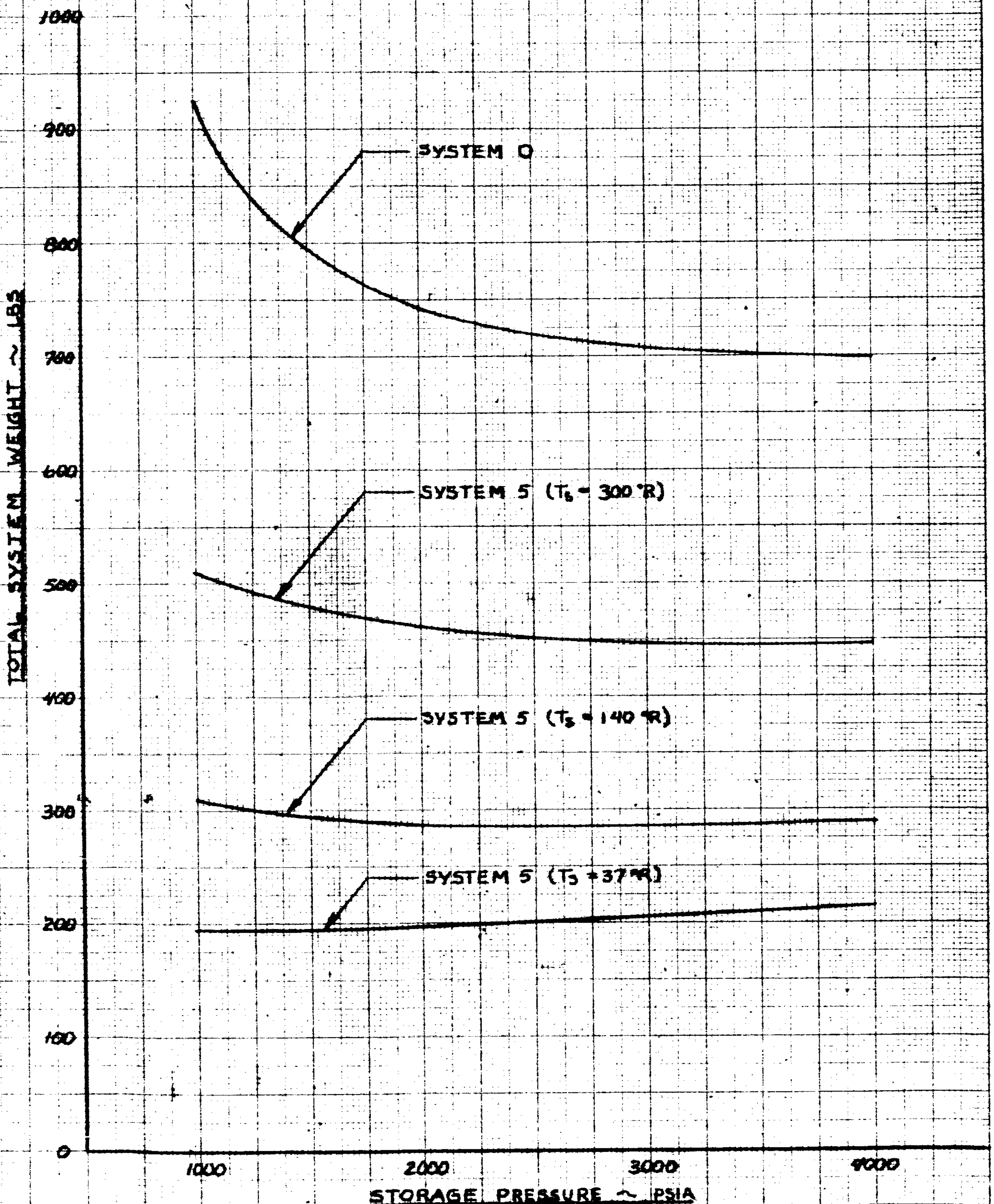
TABLE 10

SYSTEM 5

TANK INLET TEMP.	PRIMARY HELIUM STORAGE TEMP.	PRIMARY HELIUM STORAGE PRESSURE	CASCADE HELIUM STORAGE TEMP.	CASCADE HELIUM STORAGE PRESSURE	PRIMARY HELIUM EXPELLED	PRIMARY STORAGE CON. WEIGHT	CASCADE HELIUM LOADED & CASCADE STORAGE CON-TAINER WEIGHT	TOTAL WEIGHT OF VALVES	HEAT EX-CHANGER WEIGHT	TOTAL SYSTEM WEIGHT
515°R	37°R	1000 psia	530°R	4000 psia	64.2 lbs	21.6 lbs	38.8 lbs	36.0 lbs	34.0 lbs	194.6 lbs
515°R	37°R	2000 psia	530°R	4000 psia	64.2 lbs	31.0 lbs	32.2 lbs	36.0 lbs	34.0 lbs	197.4 lbs
515°R	37°R	3000 psia	530°R	4000 psia	64.2 lbs	40.2 lbs	30.6 lbs	36.0 lbs	34.0 lbs	205.0 lbs
515°R	37°R	4000 psia	530°R	4000 psia	64.2 lbs	49.0 lbs	31.1 lbs	36.0 lbs	34.0 lbs	214.3 lbs
515°R	140°R	1000 psia	530°R	4000 psia	64.2 lbs	80.8 lbs	97.0 lbs	36.0 lbs	33.0 lbs	311.0 lbs
515°R	140°R	2000 psia	530°R	4000 psia	64.2 lbs	90.1 lbs	63.2 lbs	36.0 lbs	33.0 lbs	286.5 lbs
515°R	140°R	3000 psia	530°R	4000 psia	64.2 lbs	99.2 lbs	52.0 lbs	36.0 lbs	33.0 lbs	284.4 lbs
515°R	140°R	4000 psia	530°R	4000 psia	64.2 lbs	108.3 lbs	47.0 lbs	36.0 lbs	33.0 lbs	288.5 lbs
515°R	300°R	1000 psia	530°R	4000 psia	64.2 lbs	211.5 lbs	166.0 lbs	36.0 lbs	29.0 lbs	506.7 lbs
515°R	300°R	2000 psia	530°R	4000 psia	64.2 lbs	225.0 lbs	107.0 lbs	36.0 lbs	29.0 lbs	461.2 lbs
515°R	300°R	3000 psia	530°R	4000 psia	64.2 lbs	238.0 lbs	80.2 lbs	36.0 lbs	29.0 lbs	447.4 lbs
515°R	300°R	4000 psia	530°R	4000 psia	64.2 lbs	250.5 lbs	67.8 lbs	36.0 lbs	29.0 lbs	447.5 lbs

FIGURE 34

TOTAL SYSTEM WEIGHTS FOR SYSTEM 5



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E. SYSTEM 7

The main tank injection pressurization system applied to the Apollo system and analyzed for performance predictions is basically four injectors, one in each tank, supplied with propellant at 350 psia and 530°R and controlled by a pressure switch. Injectors in the fuel tanks are supplied with oxidizer and vice versa. When the propellant in the storage tank is at a pre-determined low level injection ceases; simultaneously, injection into the sump tank commences.

Primary aspects to be considered in the application of MTI to the system are the following:

1) pressure control,
2) propellant temperature limitations,
3) system weight,
4) reliability, and
5) Apollo system modifications.

These aspects will be compared, when applicable, to previously proven systems.

Pressure control is dependent upon injector response, reagent dead column, injected stream divergence, pressure sensing tolerance and response, and reagent flow rate. Injector response is not considered to be a problem for the MTI application. Previous testing at tank pressures of about 200 lb_f/in^2 resulted in pressure tolerances well within $\pm$ 4 lb_f/in^2. Reagent dead column is defined as the mass of reagent suspended in the tank after the injector receives the closing signal. This results in the development of an overpressure condition, but the effect is negligible until the distance

between the injector tip and propellant surface is of the order of ten feet. Then the system must be analyzed in detail to determine the extent of the problem. A related problem of injecting reagent a large distance is that of stream divergence. Stream divergence has the effect of increasing the gas temperature and yielding a more homogeneous reaction, both which increase the pressure at a higher rate. Also, stream divergence can cause the combustion zone to intercept the propellant tank walls and other structure within the tank. This effect can be diminished by proper injector design. Both the reagent dead column and stream divergence can be tolerated if the pressure sensing tolerance and response are adequate, and if the tank walls and structure are capable of withstanding high temperatures. Most pressure switches have a tolerance of 1% which in the Apollo system would mean 1.75 psi, leaving 2.25 psi to absorb the previously discussed adverse conditions.
Reagent flow rate is a parameter best determined by experimentation but an adequate rate can be determined by an existing computer program.

With the present MTI system and procedure for the Apollo application, propellant temperature increase is not significant until during the fourth burn. Propellant temperature becomes excessive after the seventh burn, this results from a relatively small propellant volume within which the MTI process is taking place. Preliminary analysis indicates maximum propellant temperature attained is 644°R in the fuel tank. Ullage gas temperature attains a maximum of approximately 1475°R and is normally 1000°R to 1150°R. These values are conceivably much higher than the design allowable for the existing Apollo propellant tankage.

System weight is the primary advantage of the MTI pressurization method. Small tanks which contain reagent, reagent pressure supply tanks, injectors and control equipment, are the primary system components. The other required additional weight is a function of the minimum propellant level. System weights are presented in Table 11.

Application of the MTI pressurization system to the Apollo would require the removal of the propellant tank stillwells and change to parallel outflow. With the present Apollo configuration the injector would necessarily be installed off-center and the stream directed to impinge as close to the stillwell as possible when the propellant is at the low level. Reasons for close impingement at the low level is to inject at the very minimum propellant volume, but the closeness is governed by the possibility of stream divergence, causing a reaction on the stillwell which would be structurally detrimental. Effects of this procedure would yield a non-homogeneous temperature distribution in the ullage gas and a circumferential temperature gradient at the wall. Pressure control should not be affected by the temperature distribution. In regard to the change to parallel outflow, this would increase the injector response and diminish the possibility of a pressure overshoot at the beginning of each burn. Series flow can be used successfully, although higher propellant residuals rèsult from this method. Modifications required for the MTI system as analyzed are removing the present pressurization tank, installing the reagent tanks and small helium reagent pressure supply tank, installing an injector

TABLE 11

MTI - ALPS
SYSTEM WEIGHTS

DESCRIPTION	Wt. Ea. Lbs	Quantity	Total Lbs
Injector Valve	7.00	4	28.00
Reagent Tank (ox. Storage)	.60	1	0.60
Reagent Tank (Fuel Storage)	.72	1	0.72
Reagent Required (oxidizer)	24.38	-	24.38
Reagent Required (fuel)	20.43	-	20.43
He Storage Tank (sphere @ 3640 psia)	1.80	1	1.80
Mass of He Required	.33	-	0.33
He Regulators	2.00	2	4.00
Pressure Switches	.50	2	1.00
Reagent Isolation Valves	4.50	4	18.00
Main Tank Pressure Relief Valves	4.00	4	16.00
*Required Oxidizer Residual	124.70	-	124.67
*Required Fuel Residual	67.68	-	67.68
TOTAL SYSTEM WEIGHT			307.61

*Required in the sump tanks only.

in each of the four propellant tanks, connect with the necessary electrical and propellant lines including controls, and installing valves between the storage and sump tanks. Logic systems would have to be revised to allow for prepressurization during the zero gravity condition before each burn, to control reagent supply and to control the switching sequence between the storage and sump tank pressurization.

The analysis of the MTI-Apollo system was accomplished primarily with the ØD038, 7094 computer program, which was developed to analyze main tank injection pressurization systems under an Air Force contract. This program predicts performance parameters such as pressure control, temperatures, combustion products, propellant contamination, injector frequency and reagent consumption.

F. SYSTEM 8

System 8 uses stored helium as the oxidizer tank pressurant, which is heated by exhaust products of a gas generator. The exhaust products are then used to pressurize the fuel tank.

The analysis techniques employed for system 8 followed closely those techniques developed for the previous systems. Helium usage (as pressurant) for the oxidizer tank was established by use of the system mathematical model, ØD041. Sizing of (and weight of) the helium storage was established by the method discussed in Section III-A of this report. (The extra helium needed to pressurize the two small vessels holding the gas generator propellant was also taken into consideration).

The pressurant for the fuel tank is the hot gas exhaust from the heat exchanger (a gas generator is the hot gas source to the heat exchanger). Thus the temperature of this hot gas made available to the tank top is dependent upon the amount of energy delivered to the helium in the heat exchanger.

The hot gas mass flow rate requirements are thus seen to be dictated by two considerations:

1) the energy needed to heat the helium in the heat exchanger, and
2) the mass flow and temperature requirements at the fuel side tank top.

Since helium mass flow requirements had been determined by the ØD041 program as functions of helium storage temperature, it was next possible to employ the gas-to-gas heat exchanger program (discussed in Section III-C of this report) to determine the exchanger size, gas generator size, and gas generator propellant consumption -- the results were again dependent upon the helium storage temperature (as well as upon the helium flow rate). Calculations were made for helium storage temperatures 37°, 140°, and 300°R. Tables of the required hot gas flow rates for helium heating considerations only could then be compiled. These rates are shown in Table 12 as well as the hot gas flow rate as dictated by fuel tank pressurant needs.

Examination of this table shows that the hot gas needs of the fuel tank always exceed that need dictated by the helium heating. A chosen convergence of both needs could be obtained by having a less efficient heat exchanger such that delivery of hot gas to the fuel tank would be at a higher temperature such that less hot gas would be needed. Inspection of the last two columns in the table illustrates this effect and it is seen that the effect is slight. The assumption was made that a hot gas bypass line around the heat exchanger would be employed in conjunction with a dummy secondary heat exchanger (perhaps a coil of tubing around a propellant feed line) so as to artificially drop the hot gas temperature to a lower level during those periods when the helium would not be flowing through the main heat exchanger. Necessary extra components were included in the weight summary to accomplish this. The system weight summary is as shown in Table 13 and Figure 35.

TABLE 12

COMPARISON OF HOT GAS REQUIREMENTS FOR SYSTEM 8

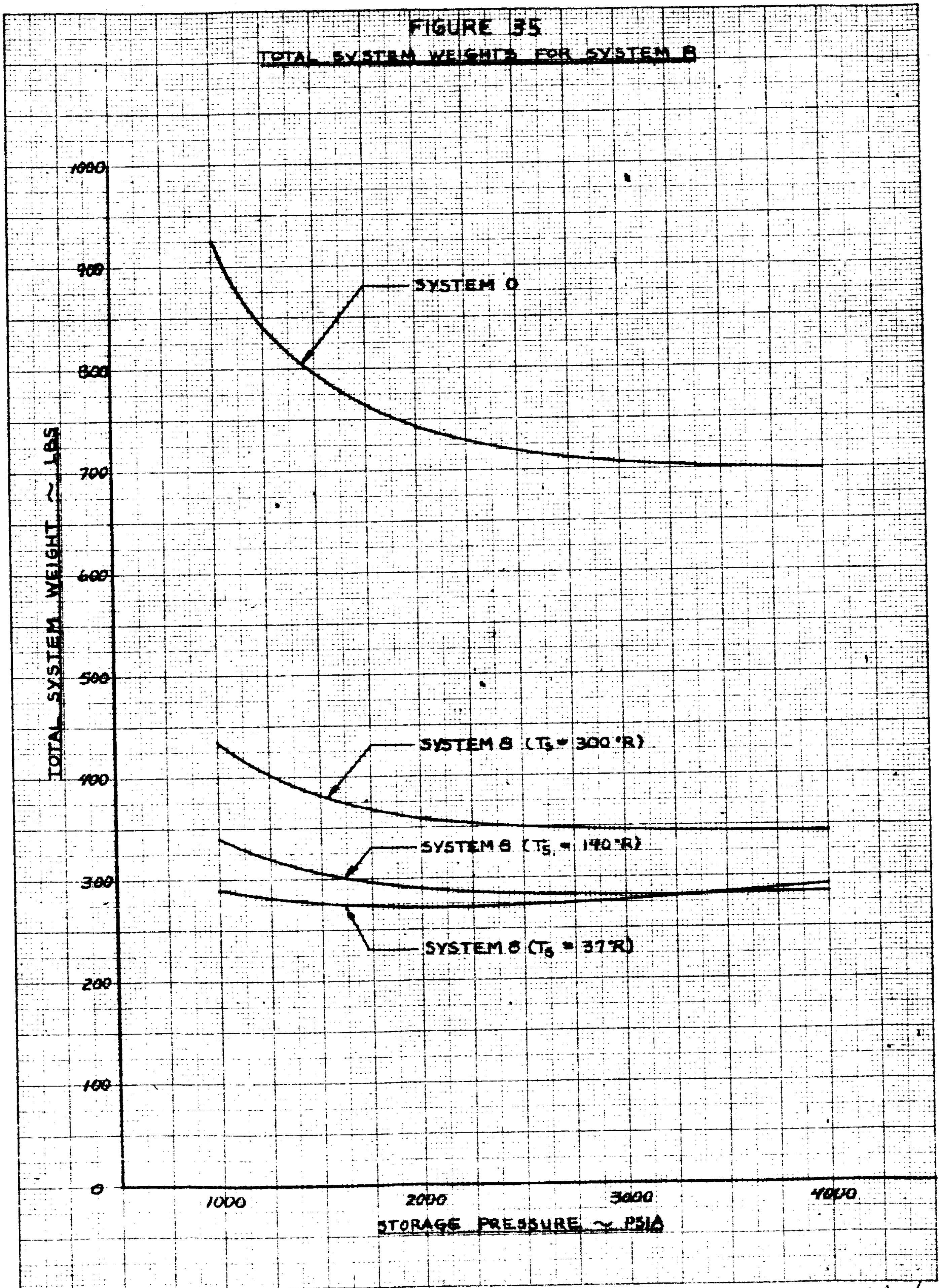
	Total hot gas required as dictated by heat exchanger considerations... helium delivered to oxidizer tank at 530°R... for helium stored at...			Total hot gas required as pressurant for fuel tank...	
	37°R	140°R	300°R	for del. at °R 620	for del. at °R 750
1st Press.	0	0	0	0.019	0.014
1st Burn	0.728	0.559	0.325	1.508	1.238
2nd Press	0	0	0	0.205	0.288
2nd Burn	0.702	0.546	0.312	1.575	1.278
3rd Press	0	0	0	0.256	0.437
3rd Burn	0.689	0.533	0.312	1.489	1.293
4th Press	0	0	0	0.323	0.511
4th Burn	21.851	16.779	9.755	50.889	45.987
5th Press	0	0	0	3.180	5.222
5th Burn	0	0	0	2.317	2.176
6th Press	0	0	0	1.550	2.894
6th Burn	0.204	0.163	0.092	0.946	0.836
7th Press	0.300	0.240	0.150	0.770	1.710
7th Burn	6.655	5.203	3.025	14.892	12.412
8th Press	0.150	0.120	0.060	1.802	3.115
8th Burn	0.090	0.070	0.042	0.228	0.218
9th Press	0.360	0.270	0.180	0.663	1.494
9th Burn	0.090	0.070	0.042	0.280	0.179
10th Press	0.360	0.300	0.180	0.342	0.664
10th Burn	0.090	0.070	0.042	0.043	0.042
TOTALS	32.269	24.923	14.517	83.277	81.975

TABLE 13

SYSTEM 8

TANK INLET TEMP.	HELIUM STORAGE TEMP.	HELIUM STORAGE PRESSURE	HELIUM LOADED AND STORAGE CONTAINER WT.	GG., A-50 CONTAINER WEIGHT	GG., NTO CONTAINER WEIGHT	GG., A-50 LOADED WEIGHT	GG., NTO LOADED WEIGHT	GAS GENERATOR WT.	HEAT EXCHANGER WT.	TOTAL WEIGHT OF VALVES	TOTAL SYSTEM WEIGHT
530°R	37°R	1000 psia	144.493 lbs	6.0 lbs	0.5 lbs	77.0 lbs	6.3 lbs	8.0 lbs	7.1 lbs	39.5 #	288.893#
530°R	37°R	2000 psia	127.124 lbs	6.0 lbs	0.5 lbs	77.0 lbs	6.3 lbs	8.0 lbs	7.1 lbs	39.5 #	271.524#
530°R	37°R	3000 psia	133.377 lbs	6.0 lbs	0.5 lbs	77.0 lbs	6.3 lbs	8.0 lbs	7.1 lbs	39.5 #	277.777#
530°R	37°R	4000 psia	145.883 lbs	6.0 lbs	0.5 lbs	77.0 lbs	6.3 lbs	8.0 lbs	7.1 lbs	39.5 #	290.283#
530°R	140°R	1000 psia	194.829 lbs	6.0 lbs	0.5 lbs	77.0 lbs	6.3 lbs	8.0 lbs	5.7 lbs	39.5 #	337.829#
530°R	140°R	2000 psia	146.577 lbs	6.0 lbs	0.5 lbs	77.0 lbs	6.3 lbs	8.0 lbs	5.7 lbs	39.5 #	289.577#
530°R	140°R	3000 psia	139.699 lbs	6.0 lbs	0.5 lbs	77.0 lbs	6.3 lbs	8.0 lbs	5.7 lbs	39.5 #	282.699#
530°R	140°R	4000 psia	140.151 lbs	6.0 lbs	0.5 lbs	77.0 lbs	6.3 lbs	8.0 lbs	5.7 lbs	39.5 #	283.151#
530°R	300°R	1000 psia	291.643 lbs	6.0 lbs	0.5 lbs	77.0 lbs	6.3 lbs	8.0 lbs	3.7 lbs	39.5 #	432.643#
530°R	300°R	2000 psia	216.331 lbs	6.0 lbs	0.5 lbs	77.0 lbs	6.3 lbs	8.0 lbs	3.7 lbs	39.5 #	357.331#
530°R	300°R	3000 psia	206.327 lbs	6.0 lbs	0.5 lbs	77.0 lbs	6.3 lbs	8.0 lbs	3.7 lbs	39.5 #	347.327#
530°R	300°R	4000 psia	203.791 lbs	6.0 lbs	0.5 lbs	77.0 lbs	6.3 lbs	8.0 lbs	3.7 lbs	39.5 #	344.791#

NOTE: NO HELIUM PURGE FOR G.G. AND HEAT EXCHANGER INCLUDED.



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IV. COMPARISON OF CANDIDATE PRESSURIZATION SYSTEMS

The candidate pressurization systems included in the preliminary study effort (discussed in Section III of this report) were subjected to a comparison study for the purpose of selecting the three systems which were most suitable for use in the Apollo Service Propulsion System. The three selected systems were then analyzed and investigated in greater detail, as discussed in Section V. The methods used in this comparison and selection effort are discussed below.

The pre-defined technique used for the comparison required a numerical evaluation of the candidate systems, related to the results of the preliminary study effort. The numerical evaluation procedure devised was based upon a comparison of certain relative merits of each candidate system, as measured by the pertinent, common characteristics among all systems. Each characteristic was assigned a "weighting factor" which was an indication of the importance of the particular characteristic in regard to the entire evaluation. The following characteristics were considered, with the indicated weighting factors.

1) Mass (1/2),
2) Reliability (1/4),
3) Compatibility and adaptability (1/4)

A merit rating number, composed of contributions from each of the above items, was computed for each candidate system. Each constituent in the merit number was defined as a ratio so that its maximum value would be one (unity), prior to multiplication by the proportional weighting factor. Therefore, the maximum value for the total merit rating number is also one. Each contributing factor is discussed below:

1. **Mass**

Since the major objective of this contract is to develop a pressurization system which is lighter than the current Apollo SPS pressurization system, mass is the most important consideration in the merit rating evaluation. The effect of mass upon the overall merit rating was defined as the ratio of mass of the lightest system to mass of the particular candidate system.

$$(N_m)_m = \frac{\text{mass of lightest system}}{\text{mass of candidate system}}$$

The value of $(N_m)_m$ will attain one as an upper limit, and may approach zero as the lower boundary.

The values for system mass used in this evaluation are comparative rather than absolute. Sizing of systems during the preliminary study effort has been only on the basis of nominal vehicle requirements, and did not consider the effects of pressurant leakage, contingent system operation, or other design perturbations which would cause arbitrarily established variations in pressurant usage. Also, the mass of tubing, insulation, support structure, etc., has been omitted, as discussed in Section III. Such items as loaded pressurant, pressurant storage containers, valves, pressure switches, gas generators, and heat exchangers are included in system mass estimates. This applies also to the present Apollo SPS pressurization system mass figure which was used in this comparison study. Rather than use the actual system mass, the system was "re-weighed" to reflect the same criteria and technique as established above for the candidate systems. In this way, all systems can be compared in the proper relative perspective.

2. Reliability

Reliability is the second key criterion affecting pressurization system selection. Man-rating of the final selected system will be an ultimate necessity, so a favorable preliminary reliability characteristic is mandatory. It is ineffective to compare a ratio of reliability numbers directly, because variations usually occur only beyond the second significant digit. A ratio of ailure rates is a much more sensitive method of comparison. Therefore, the contribution of reliability to the merit rating number is defined as

$$(N_m)_R = \frac{1. - \text{Reliability number for present system}}{1. - \text{Reliability number for candidate system}}$$

Reliability numbers were calculated on the basis of generic failure rates established for the individual components associated with each candidate system. Environmental and mission dependent effects were considered where applicable. The reliability analysis, like the mass analysis, excluded from consideration certain items not directly related to this comparison effort. Vent-relief valves, pressurant fill valves, and filters were omitted. These components are common to all systems studied, and in general do not contribute to airborne (in-flight) system operation. Items which received attention in the reliability analysis included flow control valves, check valves, pressure switches, heat exchangers, gas generators, pressurant storage containers, lines and fittings.

3. Compatibility and Adaptability

In this evaluation, compatibility is a term used to indicate the capability of each candidate in conforming to the constraints imposed by the Apollo SPS vehicle and mission. This includes such considerations as environment, geometry, operational characteristics, and logistics. In general, compatibility is a measure of the ease and potential of making any given pressurization system an operational part of the Apollo Service Propulsion System. The adaptability portion of this criteria indicates a consideration of the degree of development problems associated with each system. A state of the art system utilizing off the shelf hardware would get a higher rating than a more advanced concept for which an extensive component development program would be required. The total contribution of compatibility and adaptability to the merit rating is

$$(N_m)_c = (X)$$

where (X) is a number chosen as follows:

(X) = 0. indicates a definite incompatibility with the SPS.

(X) = .25 indicates the candidate system could probably be used with the SPS, but extensive development would be required, along with probable changes in SPS existing design.

(X) = .50 indicates the candidate can be used with the SPS, after moderate development effort and with few concessions in existing design or operation.

(X) = .75 indicates the candidate can be used with the SPS, and requires a minimum of development effort and minimum interference with existing SPS operation.

(X) = 1.0 indicates no development effort required, and the candidate is completely compatible with existing SPS system design.

The final merit rating number is computed as the sum of each of the three contributors.

$$N_m = 1/2\ (N_m)_m + 1/4\ (N_m)_R + 1/4\ (N_m)_C$$

Those systems with the highest merit rating will be considered the candidates most promising, and the systems with the lowest rating will be considered the candidates least worthy of further consideration.

Pertinent results of the candidate pressurization system comparison study are shown in Table 14. The overall merit rating numbers of candidate systems range from a low of .5583 to a high of .8426. Merit rating of the current Apollo SPS pressurization system is .6393. The total possible range of merit rating numbers is from 0. (zero) to 1. (one). The candidates with higher merit ratings were considered to be those which are more desirable for the Apollo SPS application.

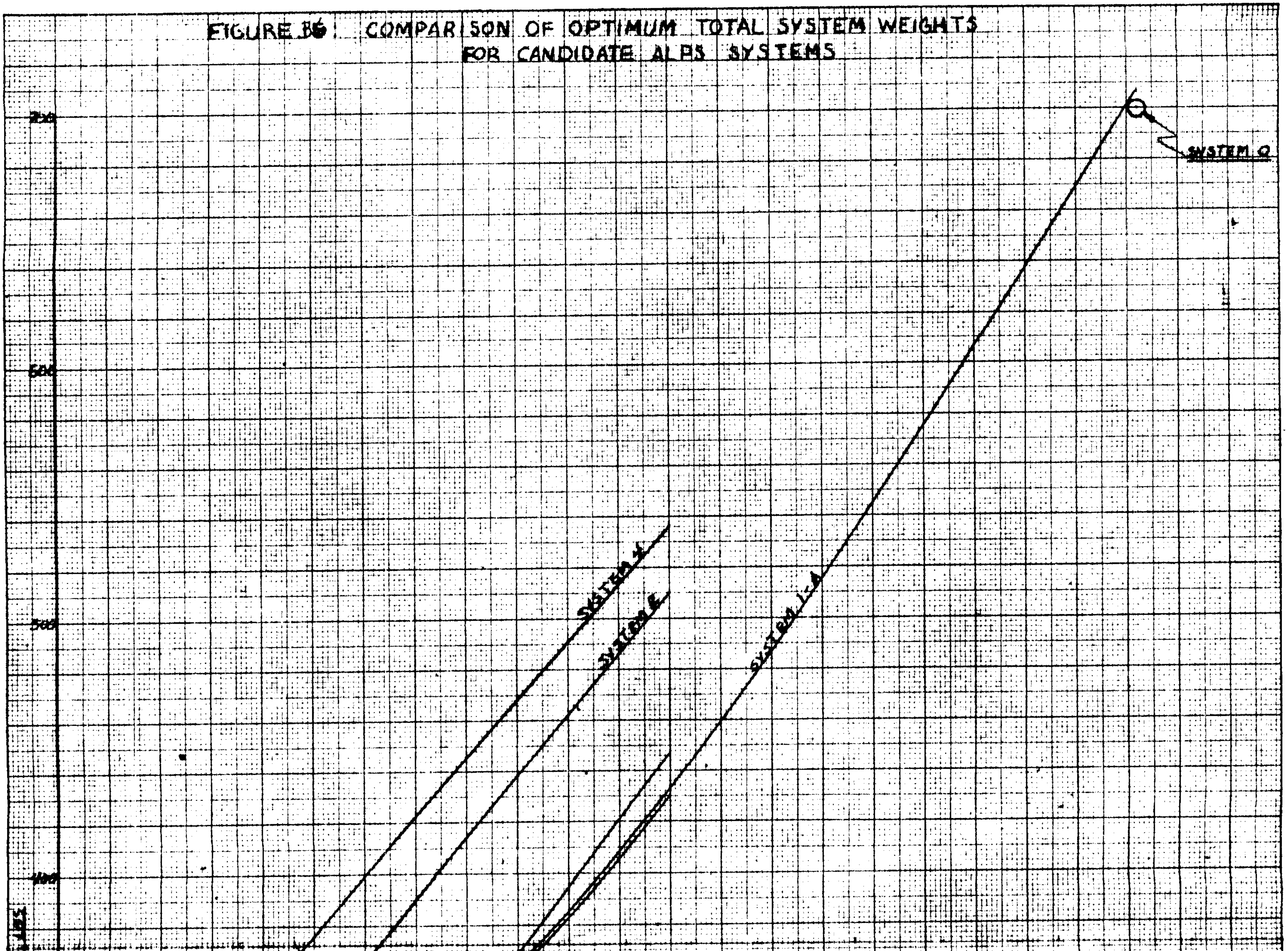
In order to more clearly illustrate the comparison of total system masses for the candidate and the current Apollo SPS pressurization systems, curves shown in Figure 36 were prepared. For the stored gas systems (1, 1A, 2, 4, 5, and 8), the curves represent total system mass as a function of initial storage temperature - at the optimum storage pressure. Systems 0 and 7 are shown as point values, since they operate only at ambient temperature.

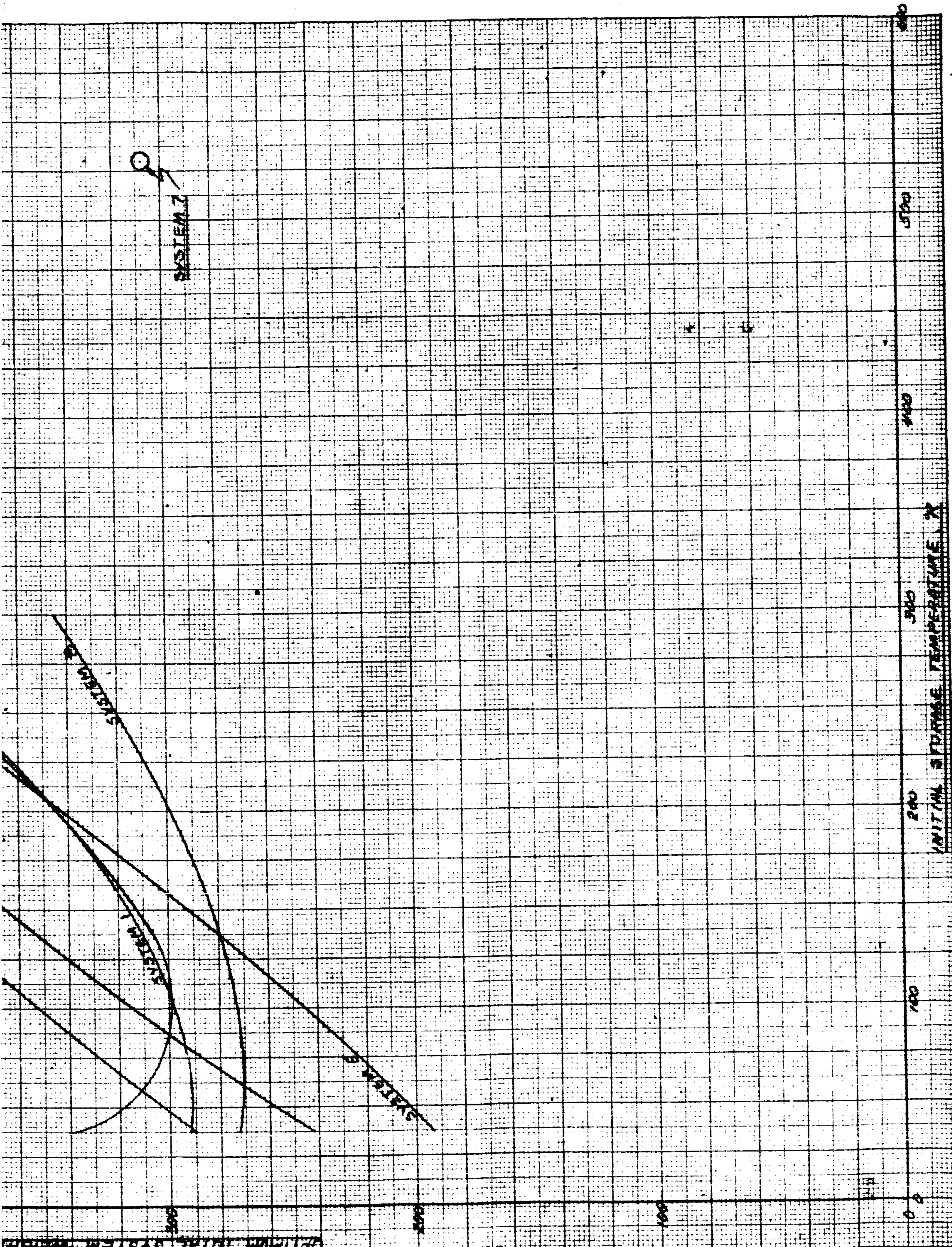
TABLE 14

MERIT RATING SUMMARY

SYSTEM	MASS (LB_m)	$(N_m)_m$	RELIABILITY	$(N_m)_R$	$(N_m)_c$	MERIT RATING
0	700	.2786	.999451	1.0000	1.00	.6393
1	291	.6701	.999374	.8770	.90	.7794
1A	298	.6544	.999373	.8756	.80	.7461
2	242	.8058	.999142	.6399	.50	.6879
4	290	.6724	.999076	.5942	.45	.5973
5	195	1.0000	.999181	.6703	.70	.8426
7	308	.6331	.999284	.7668	.20	.5583
8	272	.7169	.999268	.7500	.60	.6960

FIGURE 36. COMPARISON OF OPTIMUM TOTAL SYSTEM WEIGHTS FOR CANDIDATE ALPS SYSTEMS




SYSTEM 7
SYSTEM 1
INITIAL STORAGE TEMPERATURE, K
OPTIMUM TOTAL SYSTEM WEIGHT

The relative ranking of the candidate systems (present Apollo system included according to numerical merit rating is given in Table 15. System 5 has the highest overall rating, by a significant margin. As noted in Figure 36 it also is potentially the lightest of all candidate systems. System 1 ranks second, with system 1A a close third.

Systems 1 and 1A are very similar in concept. From Figure 36, it is noted that these two systems are also extremely comparable in mass. However, system 1A depends upon two separate working fluids, whereas system 1 uses only helium. Therefore, system 1A was excluded from further consideration because of complexity considerations.

Systems 5, 1, and 8 then, were the three top candidates in the numerical evaluation. It is also noted (Figure 36) that those three systems have the most optimum weight saving potential over the widest temperature range of all candidate systems.

Based upon this comparison, systems 5, 1, and 8 were selected as candidates for more detailed design and analysis.

Table 15 Candidate System Ranking Based Upon Preliminary Analysis and Evaluation

Rank	System	Merit Rating
1	5	.8426
2	1	.7794
3	1A	.7461
4	8	.6960
5	2	.6879
6	0	.6393
7	4	.5973
8	7	.5583

V. DETAILED DESIGN, ANALYSIS, AND EVALUATION OF PRESSURIZATION SYSTEMS SELECTED FOR CONCENTRATED STUDY

The three candidate pressurization systems selected for concentrated study were defined and evaluated in detail to determine the advantages and disadvantages of each system. The results of the detailed evaluation were employed in a comprehensive comparison of the candidate systems. From this comparison, the system offering the greatest overall advantages for use in the Apollo Service Propulsion System was selected. The methods and criteria employed in the system evaluation and the results obtained are discussed in detail in this section.

The detailed studies are discussed in relation to the following major categories.

A. Additional Study and Refinement of System Concepts.

B. Problem Area Investigation

   1. Helium Storage Tests
   2. Propellant Feedline Heat Exchanger Tests
   3. Pulsed Mode Pressurization System Tests
   4. Gas Generator/Propellant Feedline Gas Cooler Tests

C. Optimum System Selection

The thermodynamic analyses used in these detailed studies utilized the same methods and models which were used for the preliminary studies, with the addition that thermal effects of the system environment are now considered. Final pressurant storage container sizing included a 5 per cent "contingency factor" to allow for leakage and loading tolerances. Also, the system mass estimated in this section reflect the effects of <u>all</u> identifiable components required for complete flight systems.

A. Additional Study and Refinement of System Concepts

One of the early accomplishments of the detailed design and analysis effort was the review of the basic concepts for the purpose of incorporating possible improvements. Several potentially attractive modifications were considered, affecting all three candidate systems. Certain of the modifications, involving systems 5 and 8, were adopted. Others were discarded as being undesirable primarily from the aspect of increasing system mass. Prospective modifications which were studied, but not used, are discussed in Appendix B.

The modifications studied which were used, are discussed briefly in the following paragraphs.

System 5

The original concept of system 5 included a flexible membrane (bladder) within the primary helium tank. The purpose of the bladder was to retain the warm cascade gas within the primary tank, so that all available energy would be utilized in heating the primary gas (and tank) rather than being allowed to escape into the propellant tanks - where it would be relatively ineffective. In removing the bladder, it was recognized that some of the warm gas would exit from the primary tank, having some detrimental effect upon system mass. However, it was determined that by suitable diffusion of the entering cascade gas, nearly perfect mixing could be attained.

The analysis of the "bladderless" primary tank was premised upon instantaneous and thorough mixing of the entering cascade gas with the resident primary tank gas. Otherwise, the mathematical model used in the thermodynamic and sizing analysis was the same as was used for the system using a bladder.

The results of the analysis are shown in Table 16 and Figure 37. These weights were generated by the same rules as used during the preliminary study, and are to be interpreted as comparative figures rather than absolute. The results show an increase in the minimum system weight for all three storage temperatures. The increase in minimum weight for 300°R, 140°R, and 37°R was 4.2%, 9.2%, and 16.0%, respectively. Figure also shows that the minimum weight points were shifted toward slightly higher storage pressure, although the effect is barely discernible.

In consideration of the bladderless version of system 5, the following observations were immediately apparent:

1. Development cost and schedule uncertainties would necessarily be less than for the original concept of system 5 (with a primary tank bladder)
2. Reliability is higher than for the original concept of system 5 and
3. System weight is slightly higher than for the original version.

The only disadvantage in removing the bladder from system 5 was the increase in system weight. As noted in Figure 37, the weight increase at an initial storage pressure of 4000 psia amounts to less than 25 lb_m. The development and reliability factors were recognized as being more important than the system weight difference; therefore, the use of a bladder in the primary helium storage tank was not considered in further examination of System 5.

System 8

System 8 has been modified in two aspects since the preliminary studies were completed. The first modification was the replacement of the original gas-to-gas heat exchanger (for heating the helium and cooling

Table 16 - Cascade Pressurization System Comparison

Primary Container Storage Temperature (°R)	Primary Container Storage Pressure (PSIA)	Cascade System with a Bladder		Cascade System Without a Bladder	
		Cascade Container Subsystem Weight (LBS)	Total System* Weight (LBS)	Cascade Container Subsystem Weight (LBS)	Total System* Weight (LBS)
300.0	1000	166.0	506.7	222.21	562.91
	2000	107.0	461.2	136.91	491.11
	3000	80.2	447.4	104.07	471.27
	4000	67.8	447.5	86.52	466.22
140.0	1000	97.0	311.0	157.44	371.44
	2000	63.2	286.5	100.58	320.88
	3000	52.0	284.4	79.87	312.27
	4000	47.0	288.5	69.07	310.57
37.0	1000	38.8	194.6	78.36	234.16
	2000	32.2	197.4	60.86	226.06
	3000	30.6	205.5	55.12	230.02
	4000	31.1	214.3	52.20	235.40

*NOTE: The other subsystem weights and components weights of System 5 remained unchanged.

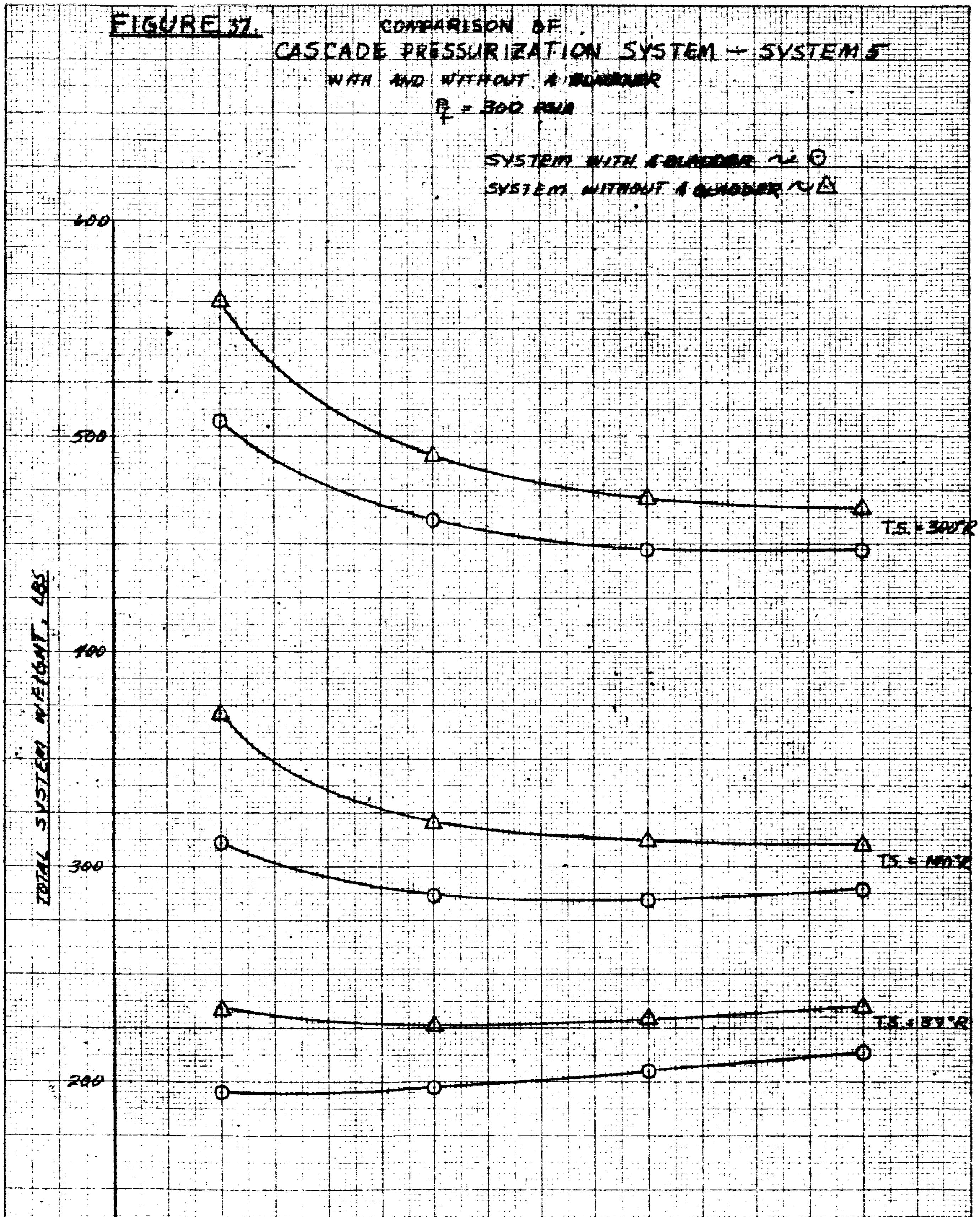


FIGURE 37.
COMPARISON OF
CASCADE PRESSURIZATION SYSTEM + SYSTEM 5
WITH AND WITHOUT A BLADDER
P_T = 300 PSIA
SYSTEM WITH A BLADDER ~ ⊙
SYSTEM WITHOUT A BLADDER ~ △
600
500
400
300
200
100
1000
2000
3000
4000
TOTAL SYSTEM WEIGHT, LBS
INITIAL STORAGE PRESSURE, PSIA
T.S. = 300°R

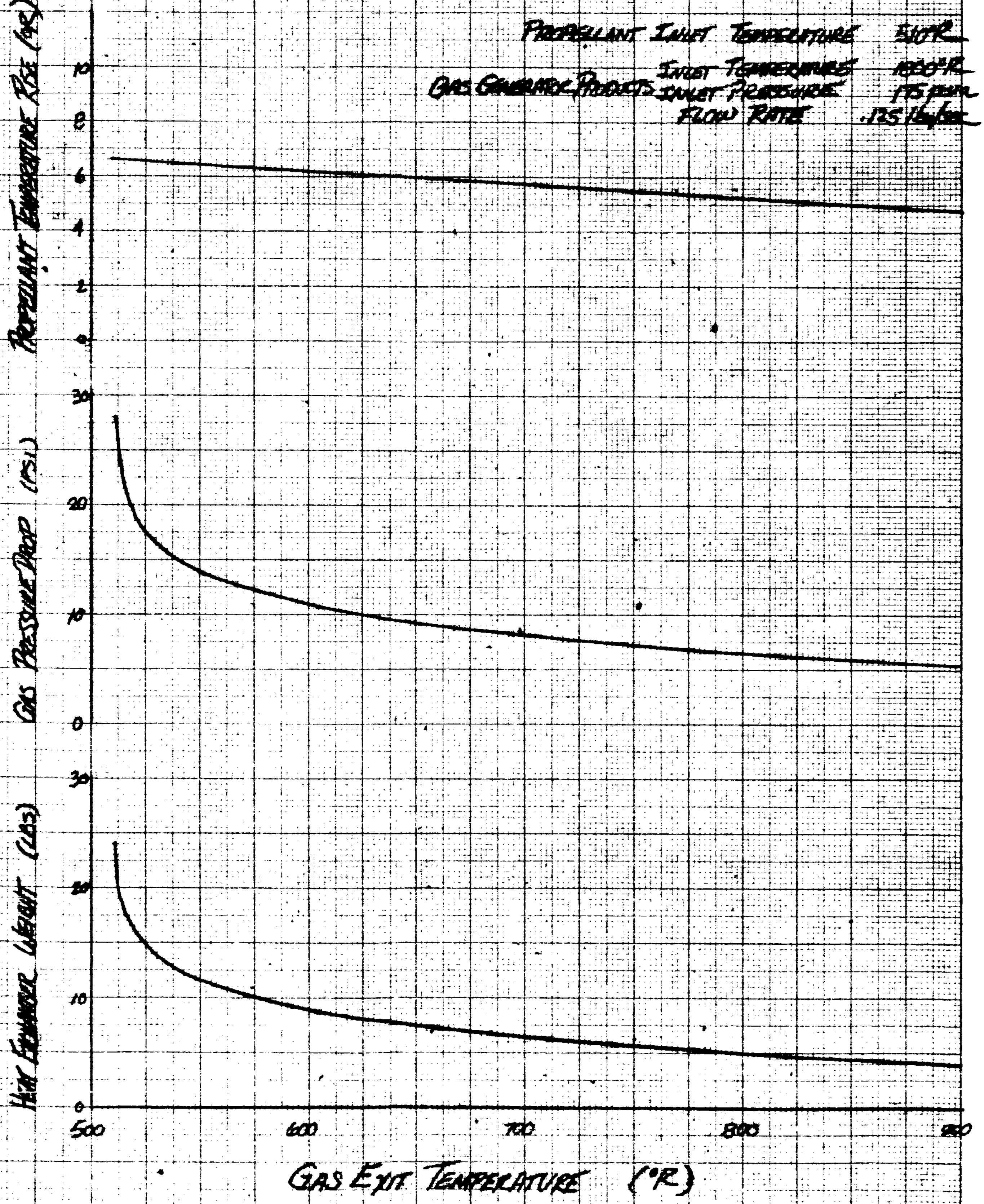
the gas generator products) with two liquid to gas propellant feedline heat exchangers. The other modification entailed replacing the bipropellant gas generator with a monopropellant unit.

In the early stages of the detailed analysis, it was found that a basic energy unbalance would prevent the use of a direct gas-to-gas heat exchanger. The energy which must be lost by the hot gas generator products in order to achieve the maximum tank entering temperature of $600^{o}R$ far exceeded the amount of energy required to heat the cold helium to that same temperature. It was then decided to analyze the use of separate propellant feedline heat exchangers - one to heat the helium to near ambient temperature, and the other to cool the hot gas generator exhaust products to an acceptable temperature ($600^{o}R$). The helium/oxidizer heat exchanger had already been analyzed in the system 1 studies (the operating parameters for the system 8 unit were identical to those for the system 1 helium/oxidizer heat exchangers). Therefore, it was only the hot gas/fuel heat exchanger which was of concern. Subsequent analysis of this unit provided data which was compatible with existing design requirements and indicated heat exchanger weights would be in the same range as weights for the helium/oxidizer units. The model used in this analysis is the same as was used for the helium propellant feedline heat exchangers (discussed in section III). The pertinent data (unit weight, gas pressure drop, and propellant temperature rise) are plotted in Figure 38, for the case where a bipropellant (N_2O_4/.5N_2H_4 - .5UDMH) gas generator is used. Figures 39 and 40 summarize the analysis for a monopropellant (N_2H_4) gas generator. These results led to the incorporation of feedline heat exchangers into the

FIGURE 38

FEED LINE HEAT EXCHANGER

A-50/N_2O_4 GAS GENERATOR PRODUCTS / PROPELLANT



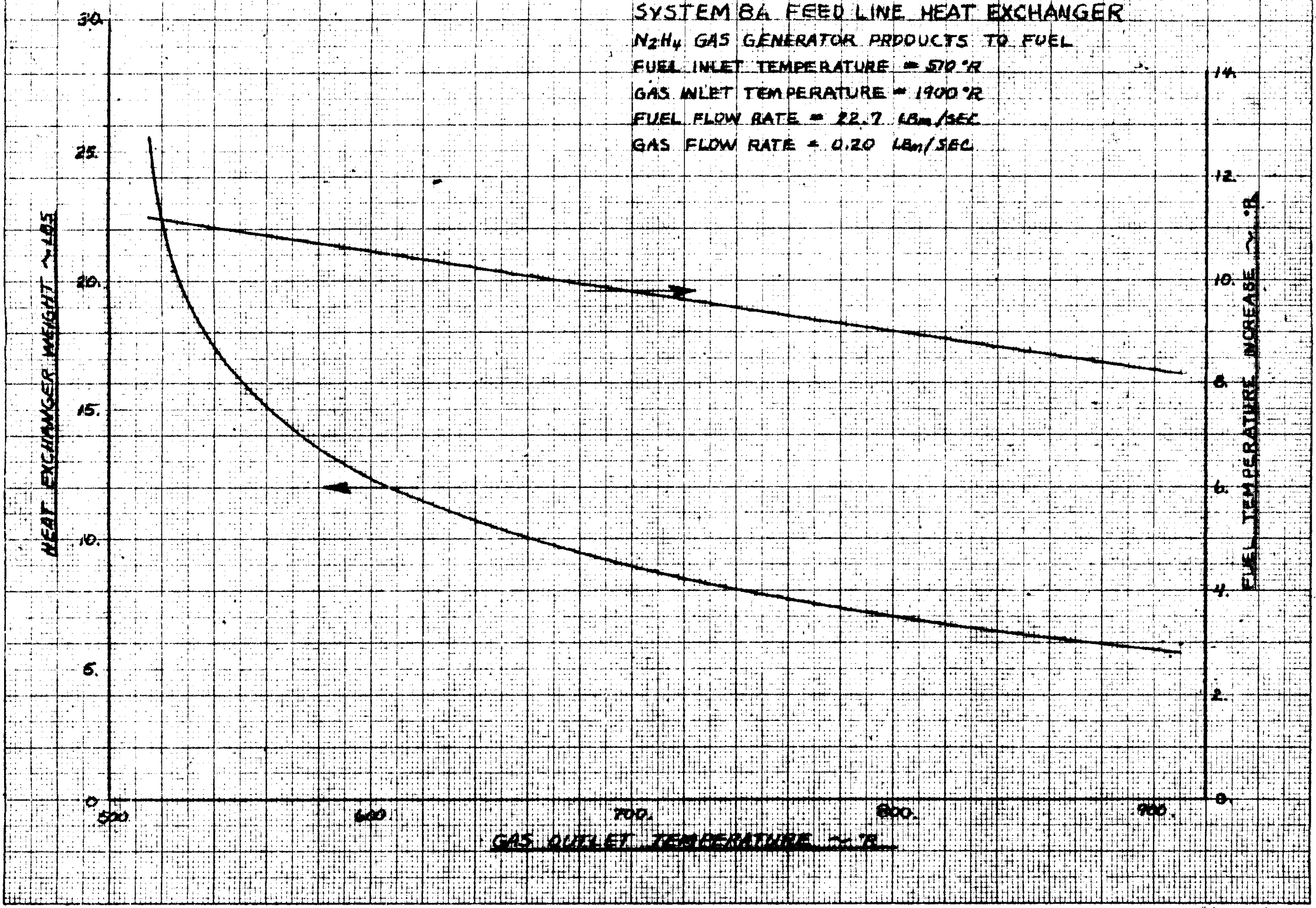


FIGURE 39

SYSTEM 8A FEED LINE HEAT EXCHANGER

N_2H_4 GAS GENERATOR PRODUCTS TO FUEL

FUEL INLET TEMPERATURE = 510 °R

GAS INLET TEMPERATURE = 1900 °R

FUEL FLOW RATE = 22.7 LBm/SEC

GAS FLOW RATE = 0.20 LBm/SEC

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FIGURE 40

SYSTEM 8A FEED LINE HEAT EXCHANGER

N_2H_4 GAS GENERATOR PRODUCTS TO FUEL
FUEL INLET TEMPERATURE = 510 °R
GAS INLET TEMPERATURE = 1900 °R
FUEL FLOW RATE = 22.7 LBM/SEC
GAS FLOW RATE = 0.20 LBM/SEC

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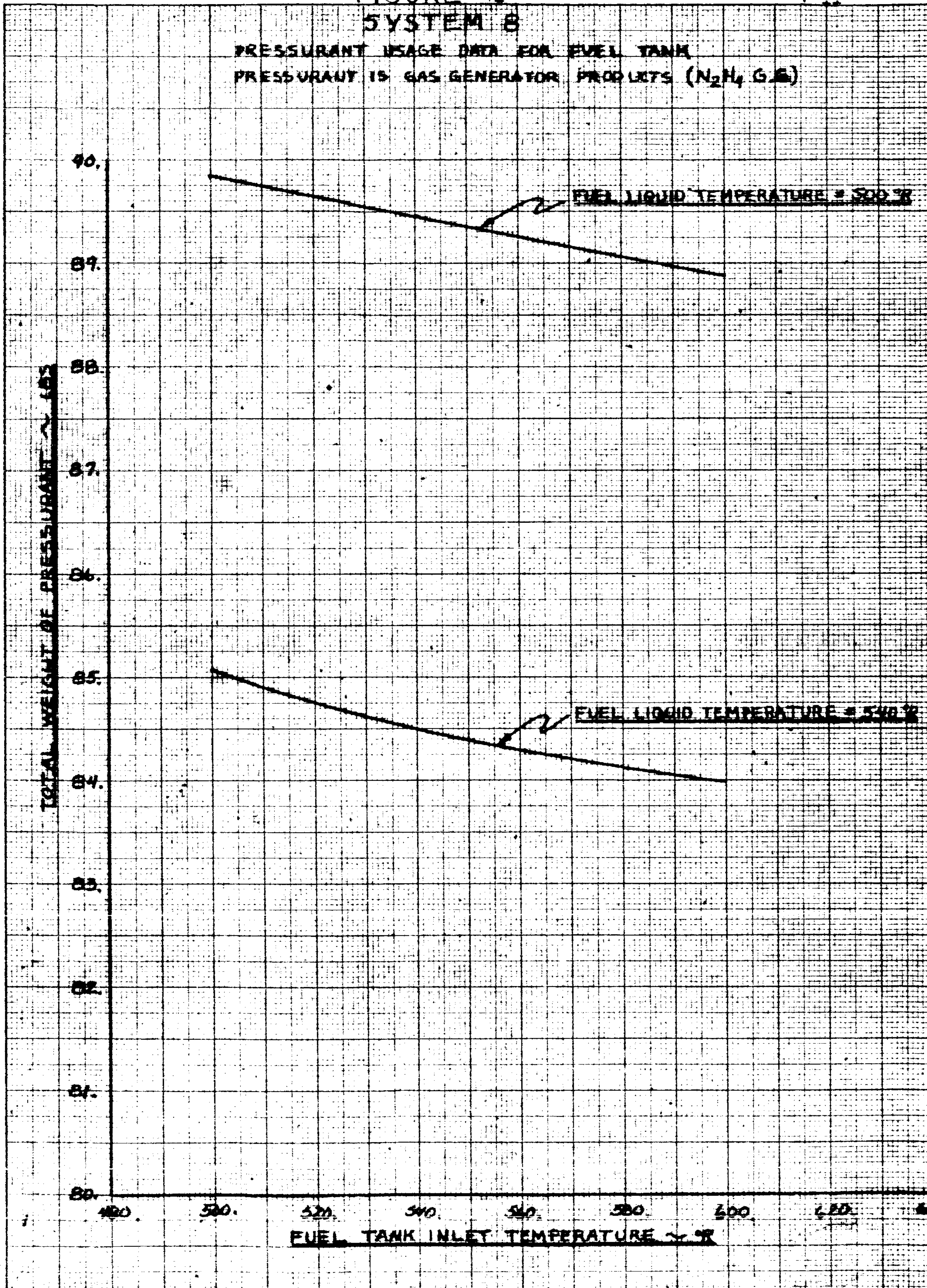
system 8 design.

The use of a hydrazine monopropellant gas generator was considered for system 8 for the following reasons.

1. System complexity (both component and operating) would be significantly reduced. Only one propellant supply subsystem would be required rather than two.
2. The combustion products are "cleaner." Only hydrogen, nitrogen, and ammonia are produced in the N_2H_4 decomposition process, all of which are compatible with the system and do not form sludge or any type of solid precipitate. The N_2O_4/.5N_2H_4 - .5UDMH reaction produces - in addition to those constituents listed above - water, vapor and various carbon compounds which are known to produce liquid and solid contaminates that are detrimental to consistant system performance.
3. The N_2H_4 monopropellant unit is capable of generating gases at lower temperatures than the bipropellant units.

Consultation with two companies prominent in the field of developing the N_2H_4 gas generator concept (Rocket Research Corporation, and Sundstrand Aviation Company) revealed that hydrazine gas generatorsare now within state-of-the-are technology, and could be used to great advantage over a bipropellant system for the Apollo SPS application. Substantiating evidence of this lies in the fact that such units are now flying on two different space vehicle systems (Mariner and Ranger). Figure 41 shows the estimated fuel tank pressurant usage requirements for N_2H_4 combustion products. These curves compare very favorably with the required usage of 89.6 lb_m

FIGURE 41



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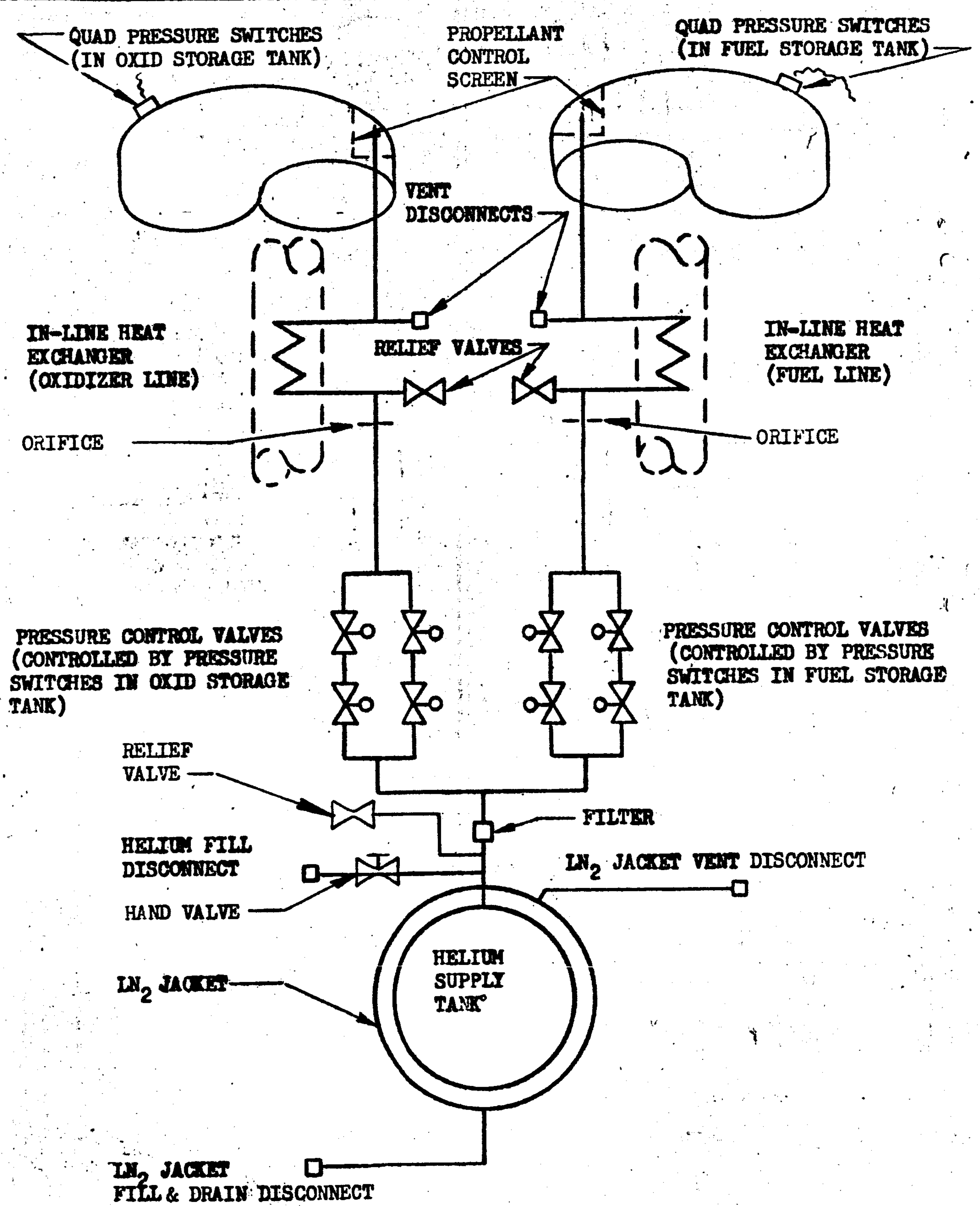
(at tank inlet temperature of 600^oR and propellant temperature of 530^oR) predicted for bipropellant gas generator products.

It was concluded that in all respects, the N_2H_4 monopropellant gas generator was more appropriate for system 8 application than the $N_2O_4/.5N_2H_4$ - .5UDMH unit.

The finalized concepts of systems 1, 5 and 8 are discussed below.

System 1

System 1 is shown schematically in Figure 42 . This system is the least complex of the three candidates, in terms of concept and operation. Helium stored at high pressure and low temperature is the pressurant. The flow of helium is controlled by solenoid valves, which are energized by pressure switches sensing propellant tank pressures. The helium is heated while enroute to the propellant tanks, by heat exchangers which utilize the propellants as heat sources. The helium storage system consists of a pressure vessel, surrounded by a lightweight, rigid jacket which is wrapped with NRC-2 superinsulation. The purpose of the jacket is to contain a coolant to maintain the pressure vessel and stored helium at the proper temperature during the pre-launch ground hold period. Any residual coolant in the jacket at launch time is vented, and does not cause a weight penalty. The propellant tank pressure switches and the solenoid valves are grouped into series-parallel units for the purpose of attaining high reliability. The orifice shown just downstream of each set of solenoid valves is used to trim the maximum helium mass flow rate for test convenience. They would not be necessary in the flight design. Propellant retention screens are used at the pressurization line outlet in each propellant



ADVANCED LIGHT-WEIGHT PRESSURIZATION SYSTEM STORED HELIUM SYSTEM PASSIVE FLOW HEATING	SYSTEM NUMBER 1

DRAWN BY: PEBingham A-65 APPROVED: [illegible]	THE MARTIN COMPANY DENVER	FIG 42 CHG B PAGE

tank to prevent liquids from backflowing through the lines. It is not necessary to prevent propellant vapors from entering the pressurization lines, since the solenoid valves preclude the possibility of mixing of the vapors from fuel and oxidizer tanks.

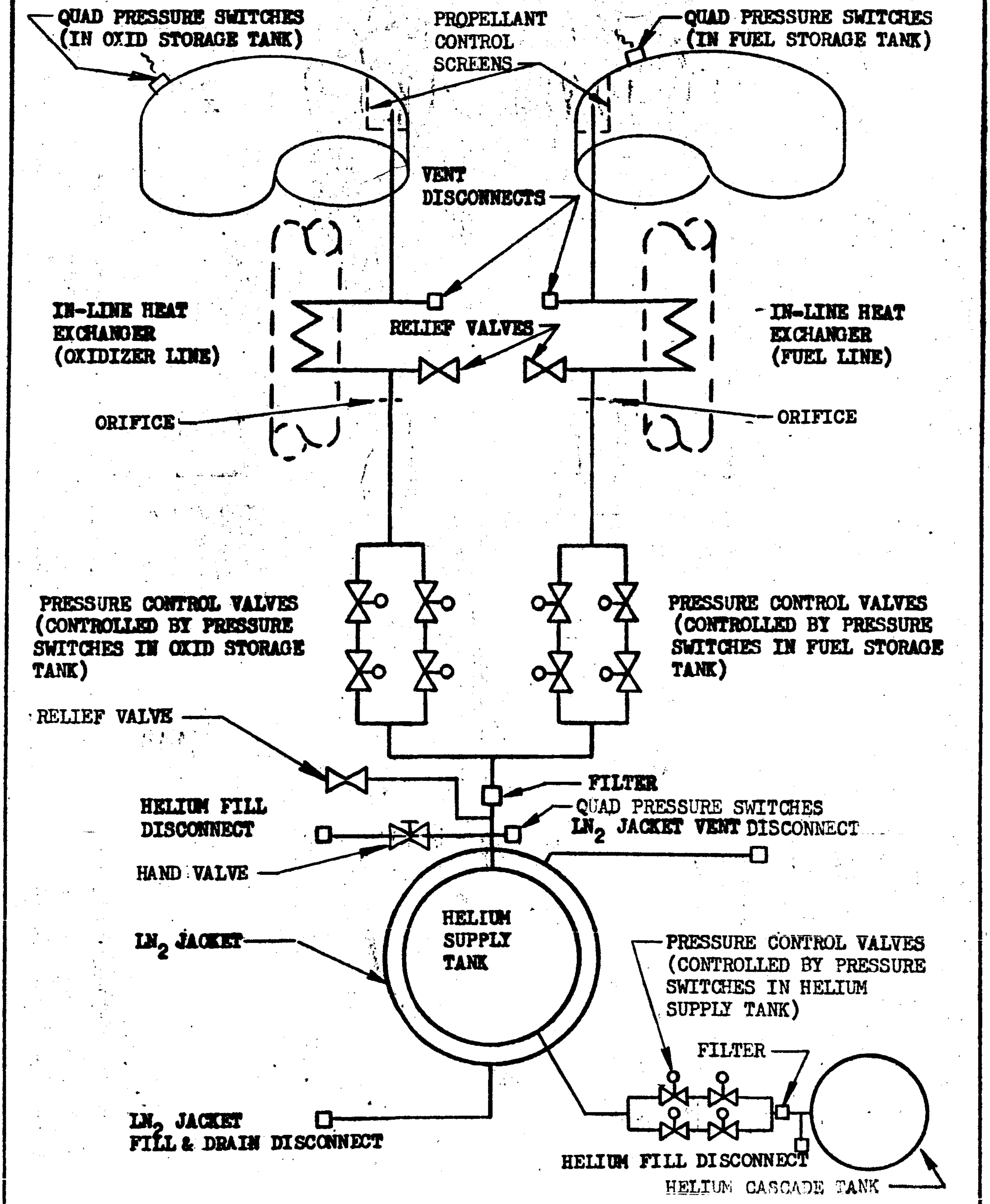
System 5

System 5 is shown in Figure 43 , and is functionally identical to System 1 with the exception that a "cascade" helium tank and associated valving and pressure switches have been added. The cascade tank contains ambient temperature helium, which is used to heat the primary storage system during latter stages of the mission. This increases the final temperature of the primary helium, thus reducing the mass of helium which must be loaded initially. The total volume of loaded helium (primary plus cascade) is also less than for system 1, which means a reduction in helium tankage mass.

Operation of the cascade arrangement is as follows. Helium for propellant tank pressurization is at all times extracted from the primary tank. When pressure within the primary tank falls below a minimum set level (in this case, 400 psia), the solenoid valves are energized admitting helium into the primary tank. Pressure in the primary tank is then controlled at 400 psia by the pressure switch-solenoid valve arrangement during the remainder of the mission. The pressure switches on the primary storage tank are arranged in series - parallel redundancy, as are the solenoid valves between cascade and primary tanks.

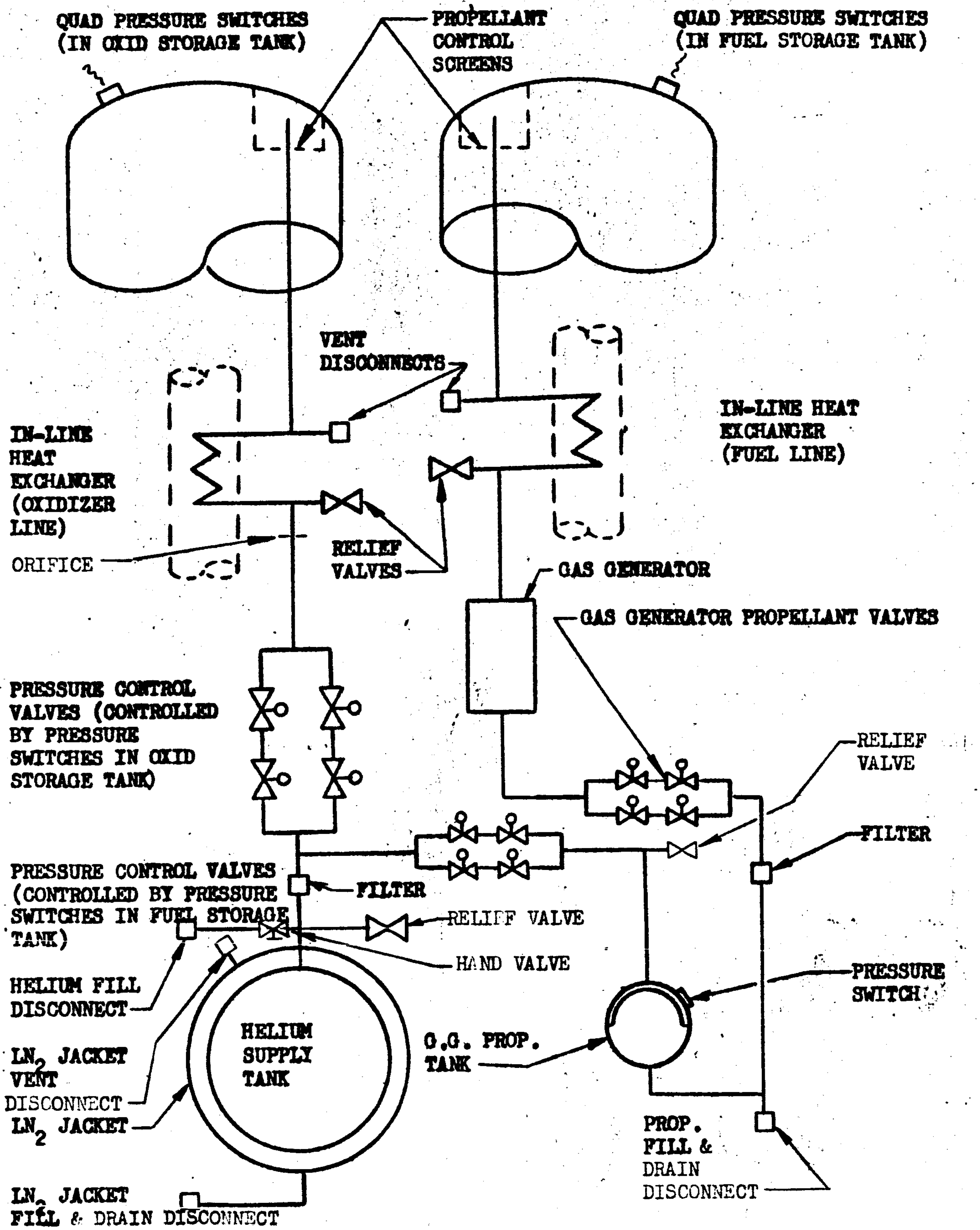
System 8

System 8 (Figure 44) consists of two separate types of pressurization systems. A cold stored helium system, identical in operation to System



ADVANCED LIGHT-WEIGHT PRESSURIZATION SYSTEM CASCADE HELIUM STORAGE SYSTEM PASSIVE FLOW HEATING	SYSTEM NUMBER 5

DRAWN BY: R.E. Burghann 4-65 APPROVED: D.M. Herman	THE MARTIN COMPANY DENVER	FIG 7B CHG B PAGE



ADVANCED LIGHT-WEIGHT PRESSURIZATION SYSTEM STORED HELIUM-PASSIVE FLOW HEATING-OXID TANK AUTOGENOUS G.G. PRODUCTS-PASSIVE FLOW COOLING-FUEL TANK	SYSTEM NUMBER 8

DRAWN BY: R.E. Kingham 4-65 APPROVED: D. Hanneman	THE MARTIN COMPANY DENVER	FIG 44 CHG A PAGE

1, pressurizes the oxidizer tank. The fuel tank is pressurized by exhaust gases from a hydrazine monopropellant gas generator. The fuel tank pressurant is composed of hydrogen, nitrogen, and ammonia. Helium from the main supply tank is used to pressurize the hydrazine tank by a quad redundant pressure switch - solenoid valve arrangement. A flexible bladder is used in the hydrazine tank to permit zero gravity operation. Hydrazine flows through a set of shut-off valves (series-parallel redundant) to the gas generator. The gas generator uses a spontaneous catalyst (Shell 405) to decompose the hydrazine at a temperature of about 1960°R. The gases are cooled in the feedline heat exchanger before entering the fuel tank. The gas generator propellant shut-off valves are operated by series-parallel redundant pressure switches on the main fuel tank.

B. Problem Area Investigation

After accessing the magnitude of the evaluation program and becoming cognizant of the data needed to conduct this detailed evaluation, it was found that further information was required. The information needed fell in two categories:

1. Performance data for components and subsystems
2. Feasibility of certain system concepts

Both analytical and experimental studies were performed to obtain the requisite information. Where applicable, experimental results were compared with analytical predictions.

The Advanced Lightweight Pressurization System (ALPS) Phase I Test Program was concerned with establishing the basic characteristics of several subsystems and components which might be used in a pressurization system for the Apollo Service Propulsion System. The general purpose of the program was to obtain empirical data on several candidate subsystems in order to establish feasibility and to validate the analytical models used for the candidate systems. The experimental investigations covered the following four basic areas of concern:

1. Thermodynamics of a cryogenic helium storage container.
2. Propellant feedline heat exchanger characteristics.
3. Operation of a solenoid valve as used in a propellant tank pressure control system.
4. Operation of a gas generator-feedline heat exchanger system as a fuel tank pressurization source.

Propellant tank venting was not investigated. Since all three candidate pressurization systems provide gas to the propellant tank inlets at near ambient[3] temperature, which is essentially equal to the ullage temperature, no requirement for venting exists. With the onset of each vehicle coast period, the bulk propellants, ullage gases, and propellant tank walls will tend to attain a condition of uniform thermal equilibrium. If the tank temperature is above ambient at this time, the subsequent cooling will cause a decrease in propellant tank pressure. If the ullage temperature is below ambient at this time, tank pressure will increase as equilibrium takes place. Propellant tank thermodynamic analysis indicates that pressurant entering temperature would have to be below about $300^{o}R$ during the burn period in order to cause tank pressures to rise above the maximum operating limit of 225 psia during coast. Since pressurant gas is injected at ambient temperature, this problem would not be encountered during normal operation.

Protection against tank overpressure due to a malfunction of either the pressure control subsystem or the in-line heat exchanger is maintained by the inclusion of combination burst disc-relief valves in all three

[3]Ambient temperature = $40^{o}F$ to $80^{o}F$

candidate systems. The burst disc provides a positive seal against leakage until the first overpressure condition occurs. This ruptures the disc, admitting gas to the downstream relief valve. If the cause for the overpressure condition is only temporary, the relief valve will reseat, providing the seal for the remainder of the mission. Typical rates of leakage through the relief valve are about 25-50 standard cc/hr (helium). Internal leak rates through a burst disc are several orders of magnitude less than 1 standard cc/hr. Therefore, it is extremely unlikely that pressure would ever build up in the space between burst disc and relief valve while the burst disc is intact. However, if this should occur, the burst disc would fail at something above the design upstream pressure. For this reason, it may prove desirable to provide a pin-hole bleed port in the assembly downstream of the burst disc.

The effects of propellant settling on the pressurization phase prior to firing were not investigated. Such investigation is not required for two reasons:

1) There is no propellant settling phase executed prior to main engine firing, since the lower part of the sump tanks are designed to retain liquid phase propellants during zero gravity coast periods; and
2) If propellant settling should become necessary prior to main engine firing, it would have negligible effect since all three candidate systems provide pressurant to the propellant tanks at essentially ambient temperature.

Item 2) insured that tank pressure will not drop below required operating limits due to cooling during coast periods.

The experimental test program is discussed below. Additional details on equipment (instrumentation), procedures, and conduction of individual tests are presented in Martin CR-64-82 (Issue 8), "Monthly Progress Report," June 1965.

1. Helium Storage Tests

Objective - The objective of this test was to acquire data on the thermodynamic characteristics of helium stored at low temperatures (about 140°R), including the effects of expansion of helium from the container and external heating of the container during simulated burn-coast periods.

Test Fixture - The test fixture, shown schematically in Figure 45 , consisted of a 4 cubic foot insulated storage sphere, an insulated vacuum tank or chamber in which the storage sphere was mounted, a radiant heater array mounted in the vacuum tank, an a LN_2/helium heat exchanger. A pictorial view of the installation is presented in Figure 46. Liquid and gaseous nitrogen and helium gas were furnished through facility lines from storage. The stainless steel sphere had an internal volume of 3.94 cubic feet and weighed 1230 pounds with the temperature rake installed. The sphere was covered with foil-backed fiberglass insulation having an installed thickness of 1/2" to 9/16". Isolation of the storage sphere mounting tabs from the vacuum chamber supports was accomplished with 15/16" thick teflon shims. The heater array consisted of twelve 200-watt strip-heater units arranged in the manner shown in Figure 45. An adjustable power supply was employed with the heaters. A 3-inch Stokes vacuum pump was connected directly to the vacuum tank following the first two tests (the capacity of the Kinney KD-30 vacuum pump

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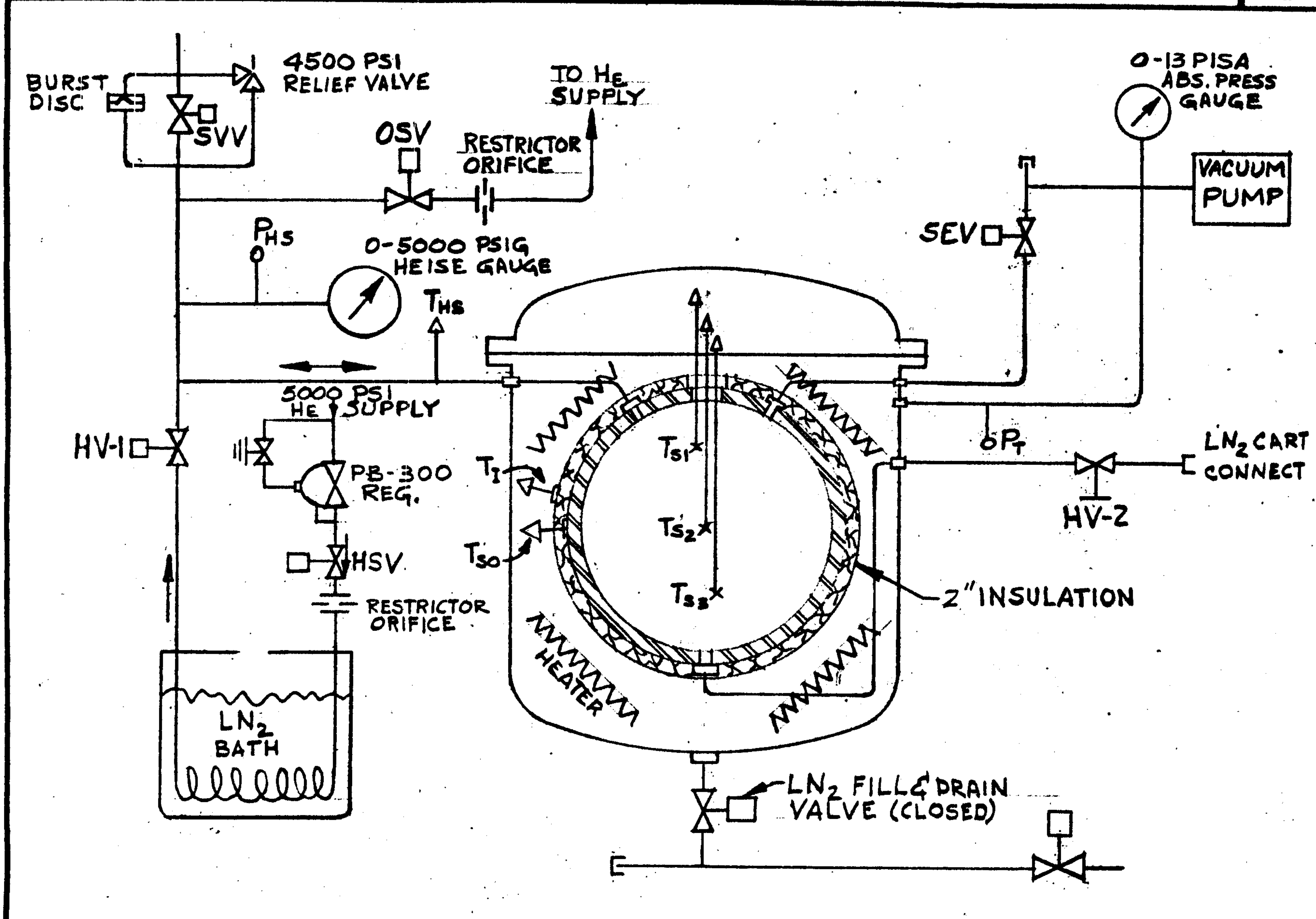
FIGURE 45 HELIUM STORAGE TEST SCHEMATIC



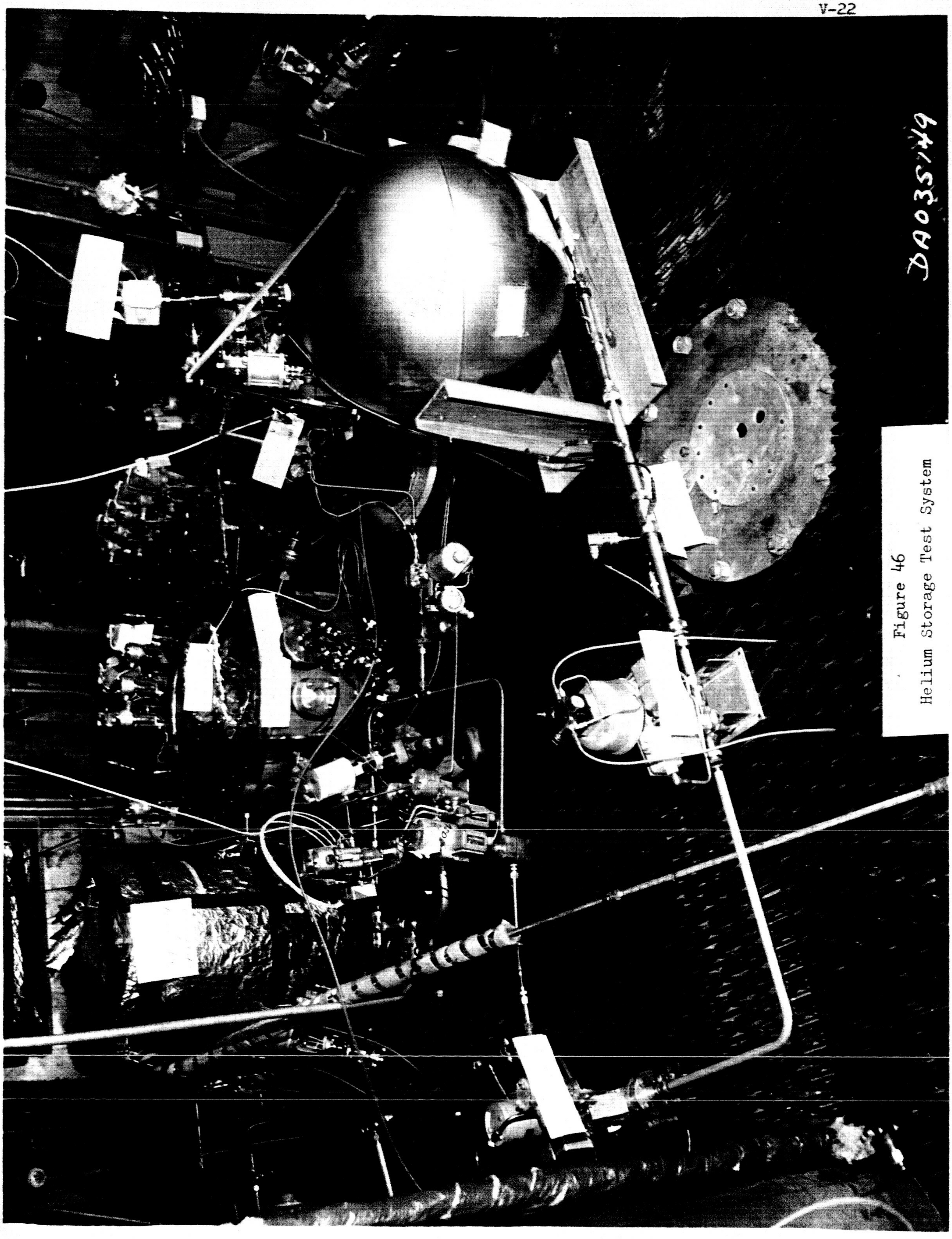


Figure 46
Helium Storage Test System

employed for the first two tests was insufficient). The helium heat exchanger consisted of sixty feet of 1/4 inch diameter by .035 inch wall thickness steel tubing immersed in liquid nitrogen. Strategically positioned thermocouples and pressure transducers were employed to monitor temperature and pressure.

Procedure - The insulated storage sphere was pre-cooled by filling it with liquid nitrogen at ambient pressure (12 psia). This condition was maintained until the sphere wall temperatures had stabilized. At this point, the temperatures sensed by the immersed gas bulk temperature thermocouple rake (T_{s_1}, T_{s_2}, T_{s_3}) were recorded as a correlation/calibration check against the LN_2 equilibrium temperature at 12 psia. The LN_2 was then drained from the sphere by pressurization with cold helium and the sphere was evacuated to scavenge the residual helium-nitrogen gas mixture. After establishment of a satisfactory vacuum in the cold sphere, helium loading was initiated. Helium was loaded through the helium heat exchanger at a rate consistent with maintaining a sphere inlet temperature of approximately 160°R. Loading was continued until the sphere was charged to 3000 psia.

The test run was initiated by setting the pressure in the vacuum tank and the electrical input to the heaters at the desired values. The sphere was maintained in a locked-up condition until an adequate gas temperature rise (15 to 20 degrees) had been observed. The gas pressure was then vented down to successively lower pressure levels (2500, 2000, 1500, 1000, 500 psia). At each of these pressure levels, the sphere was locked up for the required temperature rise period. Pertinent temperatures and pressures and the power input to the heaters were recorded continuously during the entire test run.

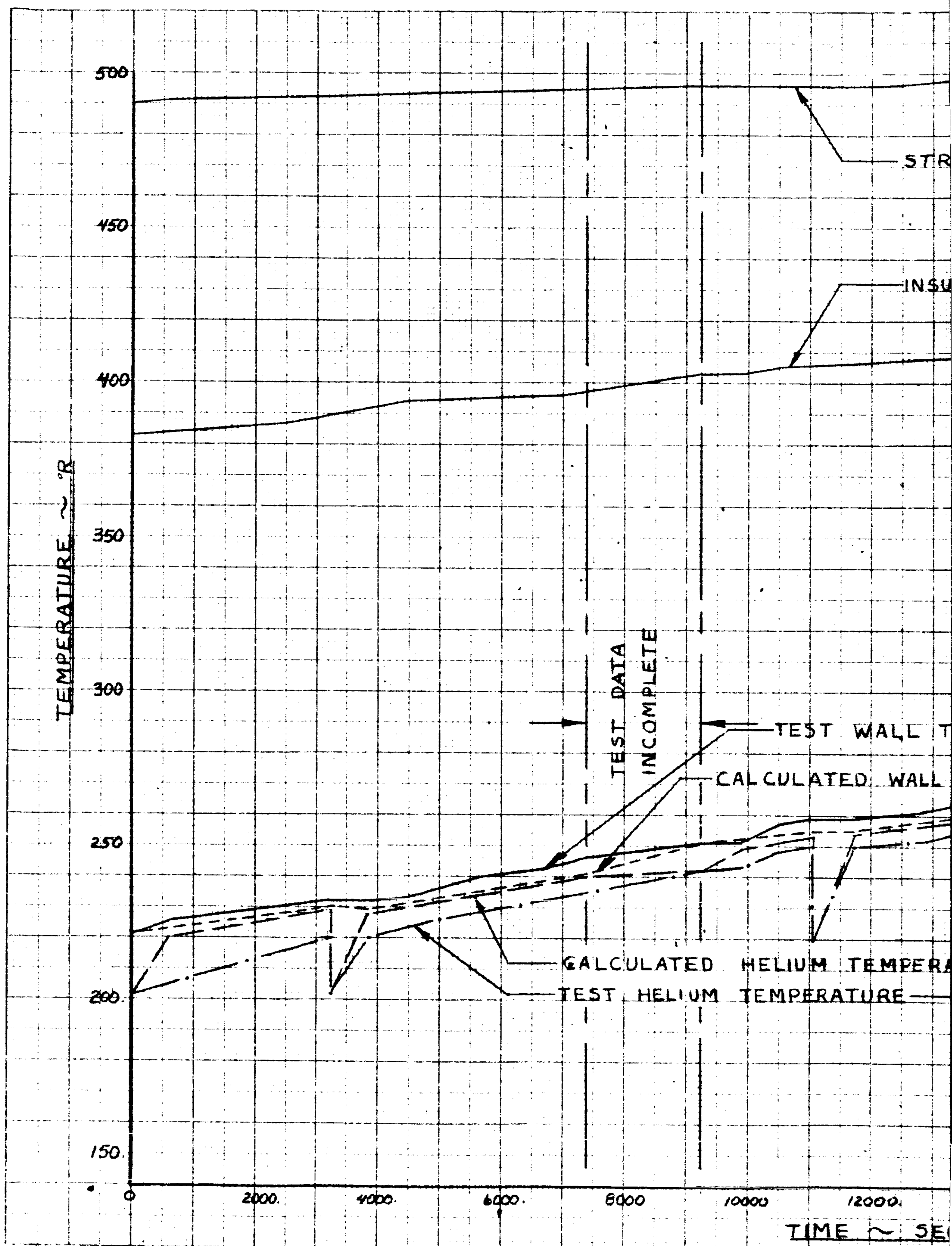
Discussion - Six runs were made; all were initiated with the helium at 3000 psi and approximately 160°R. Of the six runs, three were accomplished over the complete pressure schedule (Runs 4, 5 and 6). Valid heat transfer data were obtained during the last two runs, with the exception that a small amount of gas leakage occurred at pressure levels above 2500 psig during Runs 4 and 5. The heat flux rates used varied from those associated with free convection at ambient pressure and temperature. Runs 5 and 6, conducted with ambient temperature and pressure external environment, met the required objectives. Analysis of the results and comparison with analytical predictions are given below.

Analysis of Results - Of the six helium storage sphere tests that were executed, the last two, Runs 5 and 6, were considered successful for comparison to analytical simulation. The first four runs were not considered successful due to leakage and unpredictable heat sources.

The analytical simulation of the helium storage tests was performed with the IBM 7094 gas expansion computer program (also used for the pressurant storage analysis on the three candidate systems). One modification was made to the basic gas expansion program in order to simulate the helium storage tests. This modification was the addition of venting at given rates from an established pressure level to the next desired pressure level.

These pressure levels and flow rates were used to control the duration of each vent instead of vent time. The initial conditions for each test were input, i.e. helium pressure, helium temperature, and wall temperature, together with the external structure and insulation temperature histories, the volume, wall thickness, and sphere weight, and coast duration times, i.e. the time between each vent. Weight of helium loaded was calculated. The computer program calculated the helium pressure and temperature and the sphere wall temperature histories for the entire test. Each vent was simulated by beginning at the end of the previous coast period with helium being expelled at a given flow rate until the desired pressure was obtained. When the desired pressure was obtained, the computer program would begin the next coast period. The simulation continued until all the required ventings were completed.

The results of the evaluation and comparison of the test results with the analytical simulation are shown in Figures 47,48,49 and 50 for Tests 5 and 6, respectively. These figures show almost identical results for both tests. The calculated temperature rise rates of the helium and sphere wall temperature are very close to the temperature rise rates experienced during a large portion of both tests. The only exceptions occur at the start of the first and second coast periods for both tests. These exceptions are due to a tendency of the calculated helium temperature to approach and stay about 1 to 2 degrees below the calculated wall temperature. Due to this tendency, the calculated



STR
INSU
TEST DATA INCOMPLETE
TEST WALL T
CALCULATED WALL
CALCULATED HELIUM TEMPERA
TEST HELIUM TEMPERATURE
TEMPERATURE ~ °R
500
450
400
350
300
250
200
150
0
2000
4000
6000
8000
10000
12000
TIME ~ SE

I-26-1

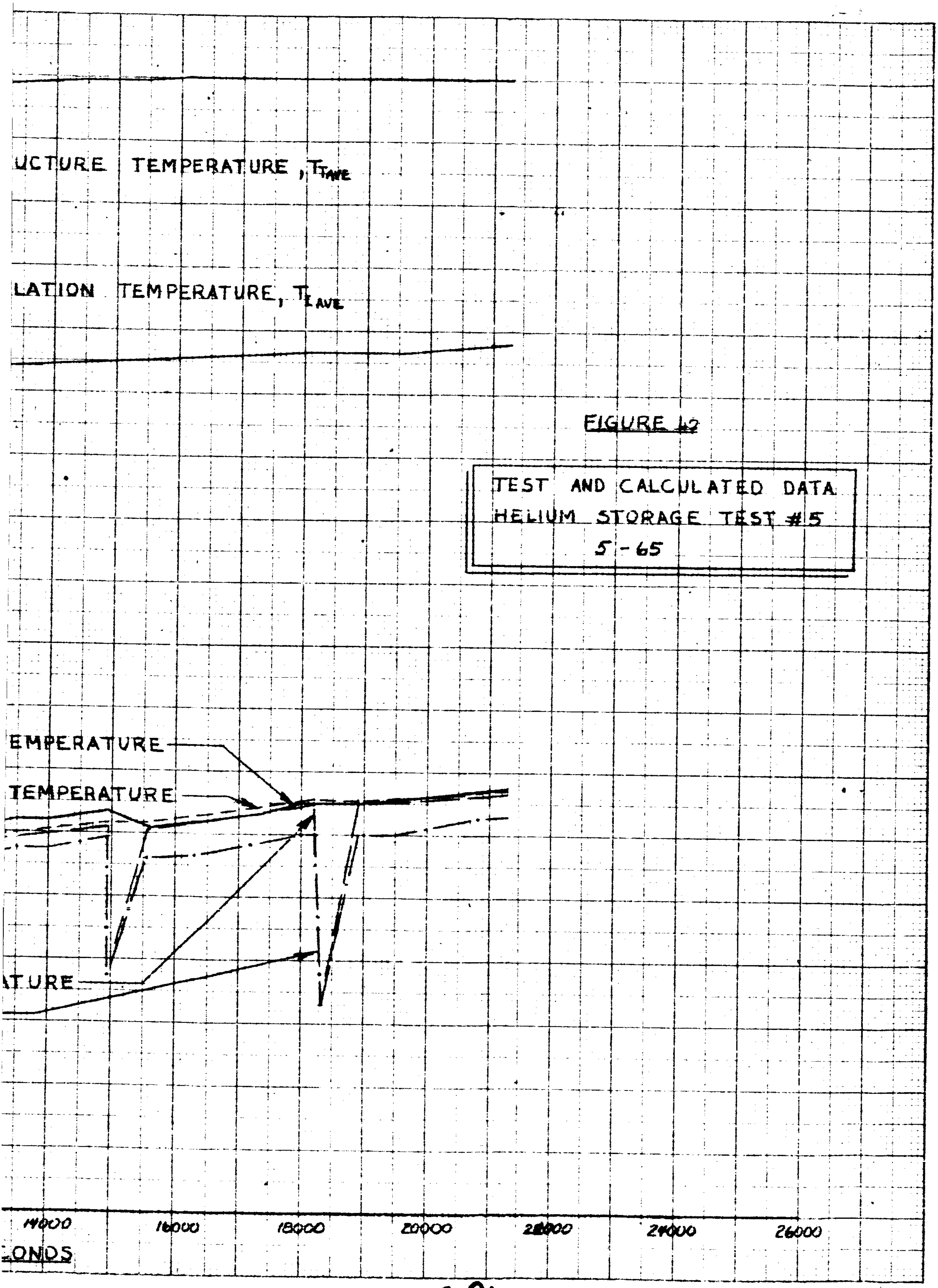
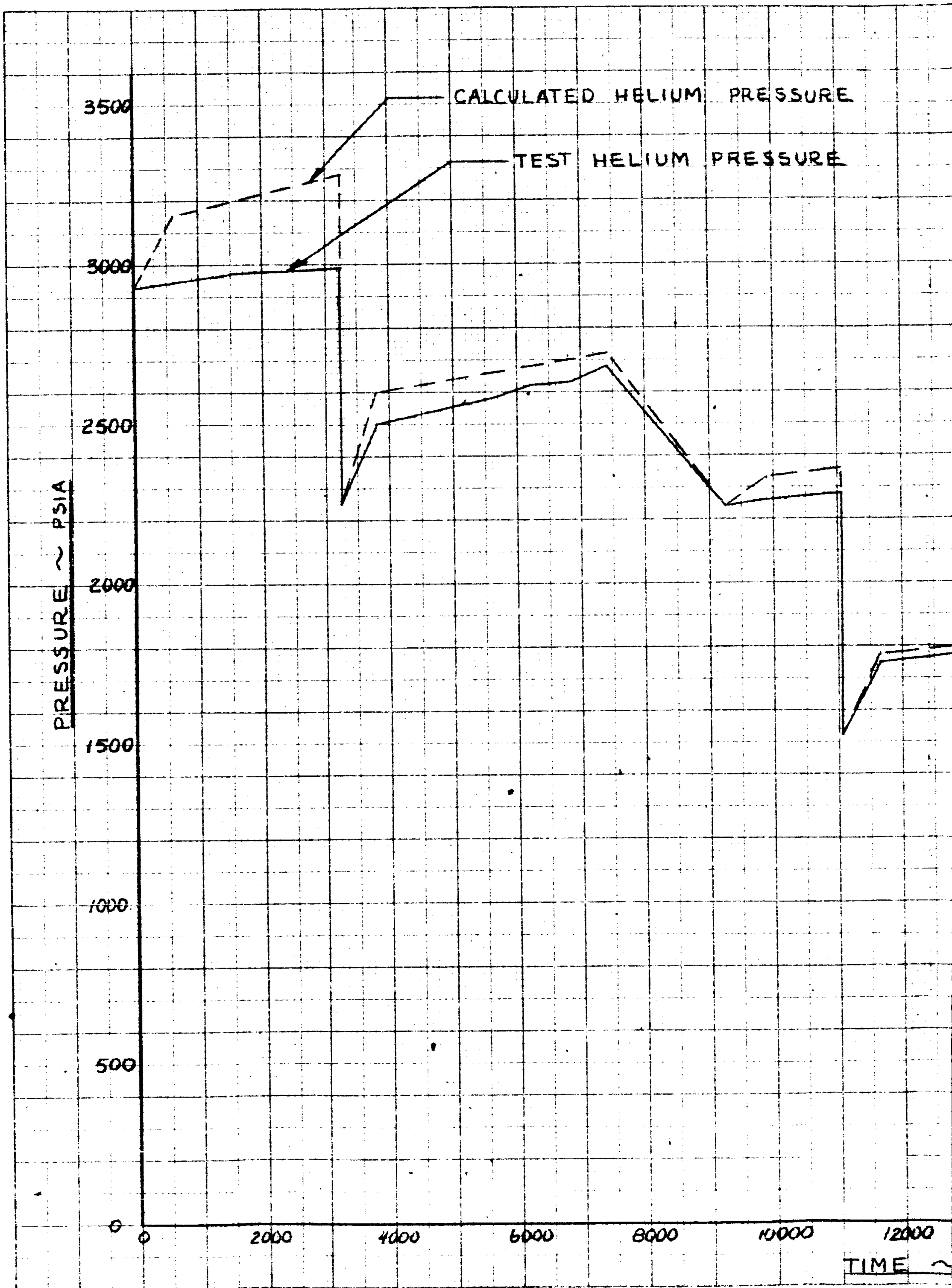


FIGURE 6

TEST AND CALCULATED DATA
HELIUM STORAGE TEST #5
5-65


CALCULATED HELIUM PRESSURE
TEST HELIUM PRESSURE
3500
3000
2500
2000
1500
1000
500
0
PRESSURE ~ PSIA
0
2000
4000
6000
8000
10000
12000
TIME ~

II-27-1

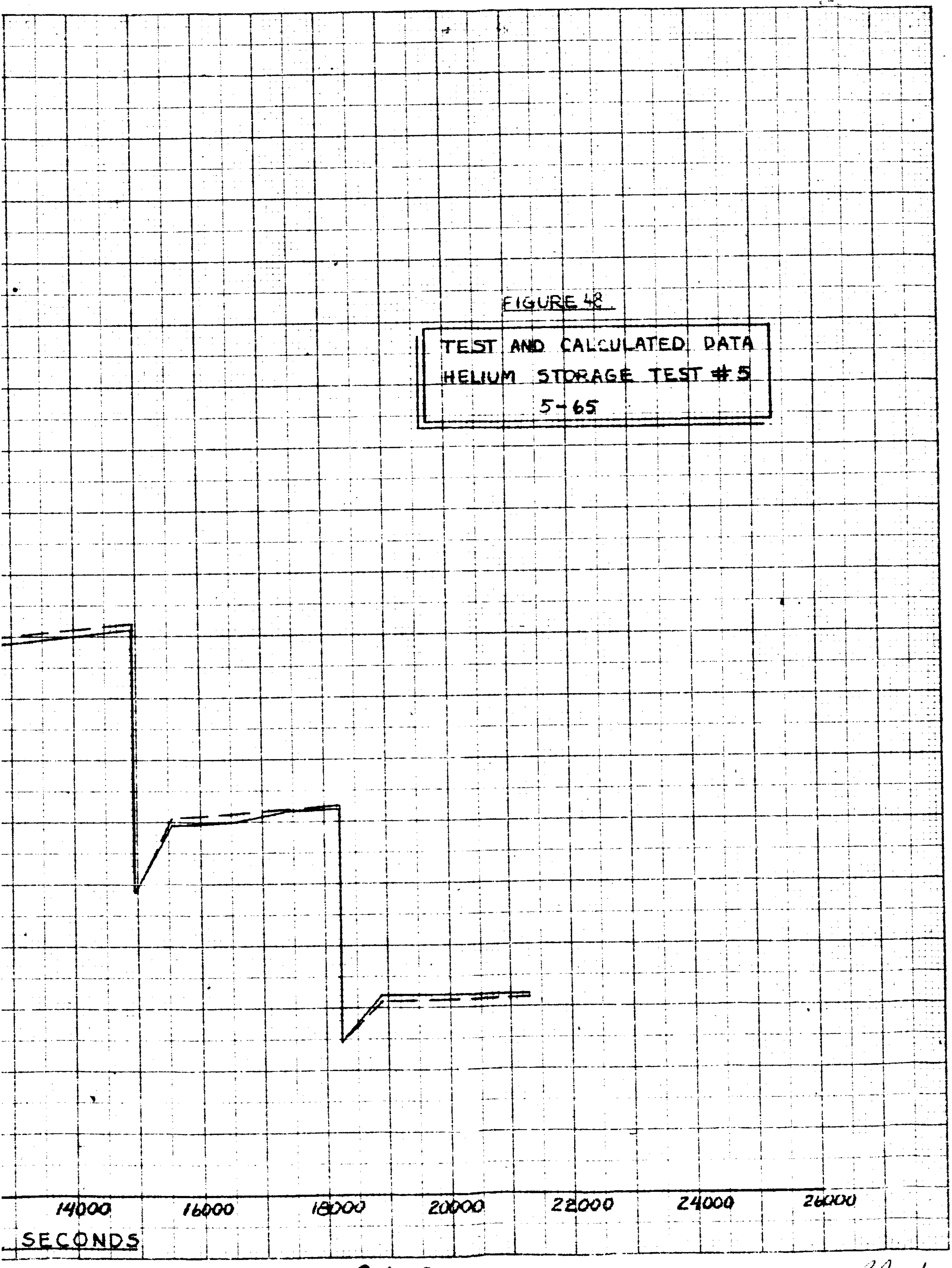


FIGURE 48
TEST AND CALCULATED DATA
HELIUM STORAGE TEST #5
5-65
14000
16000
18000
20000
22000
24000
26000
SECONDS

11-27-2

RP 7/65

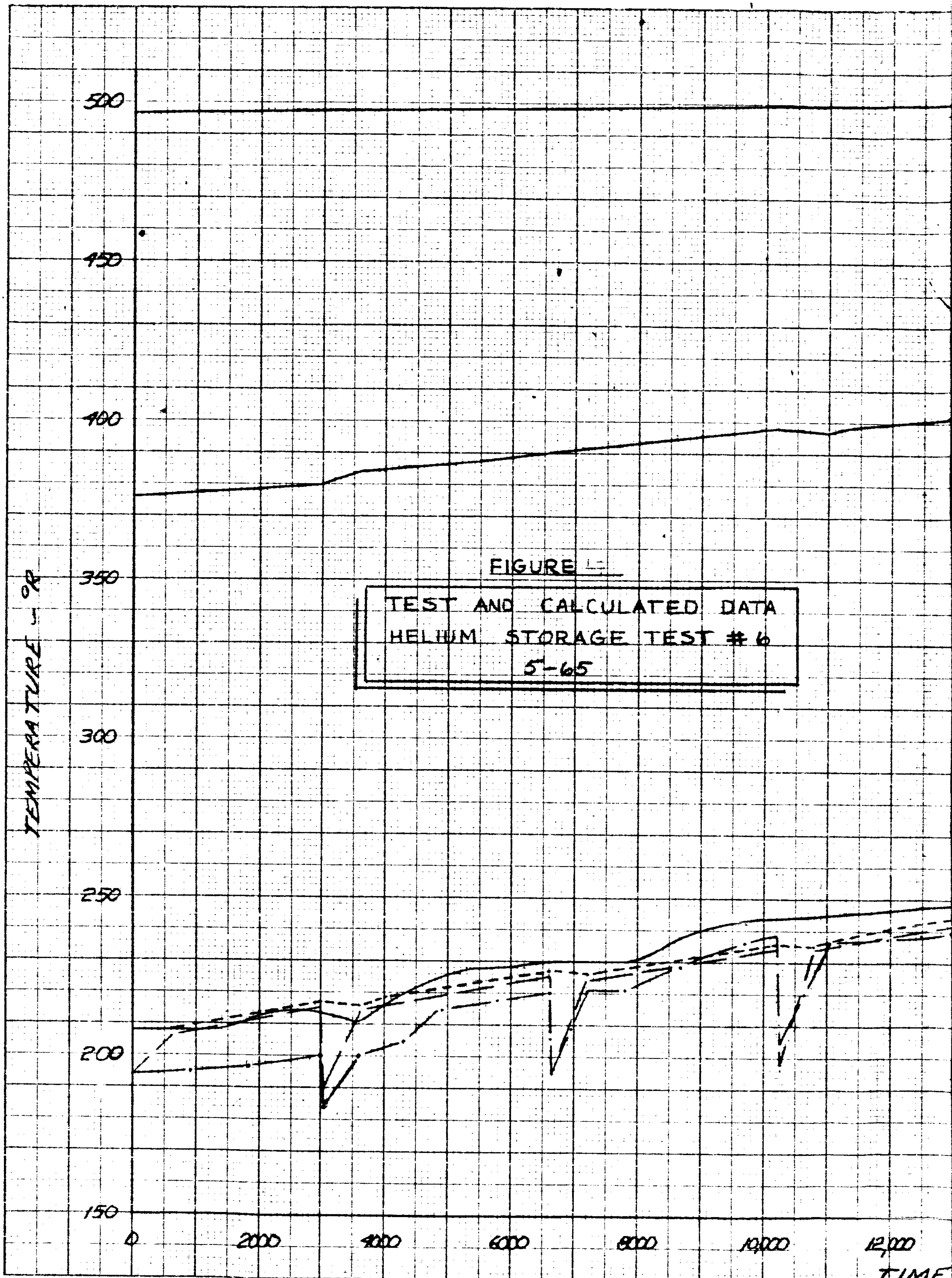
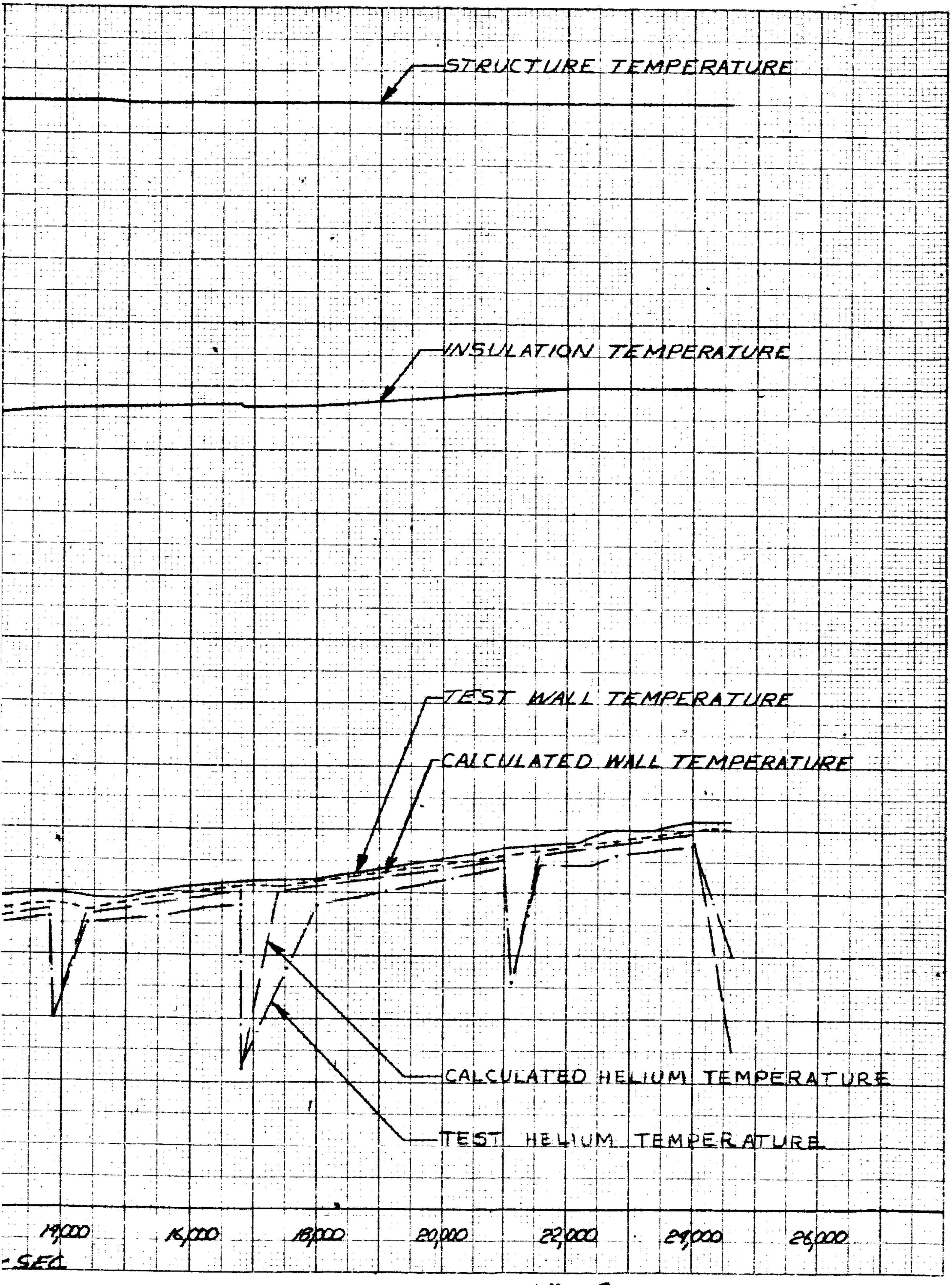
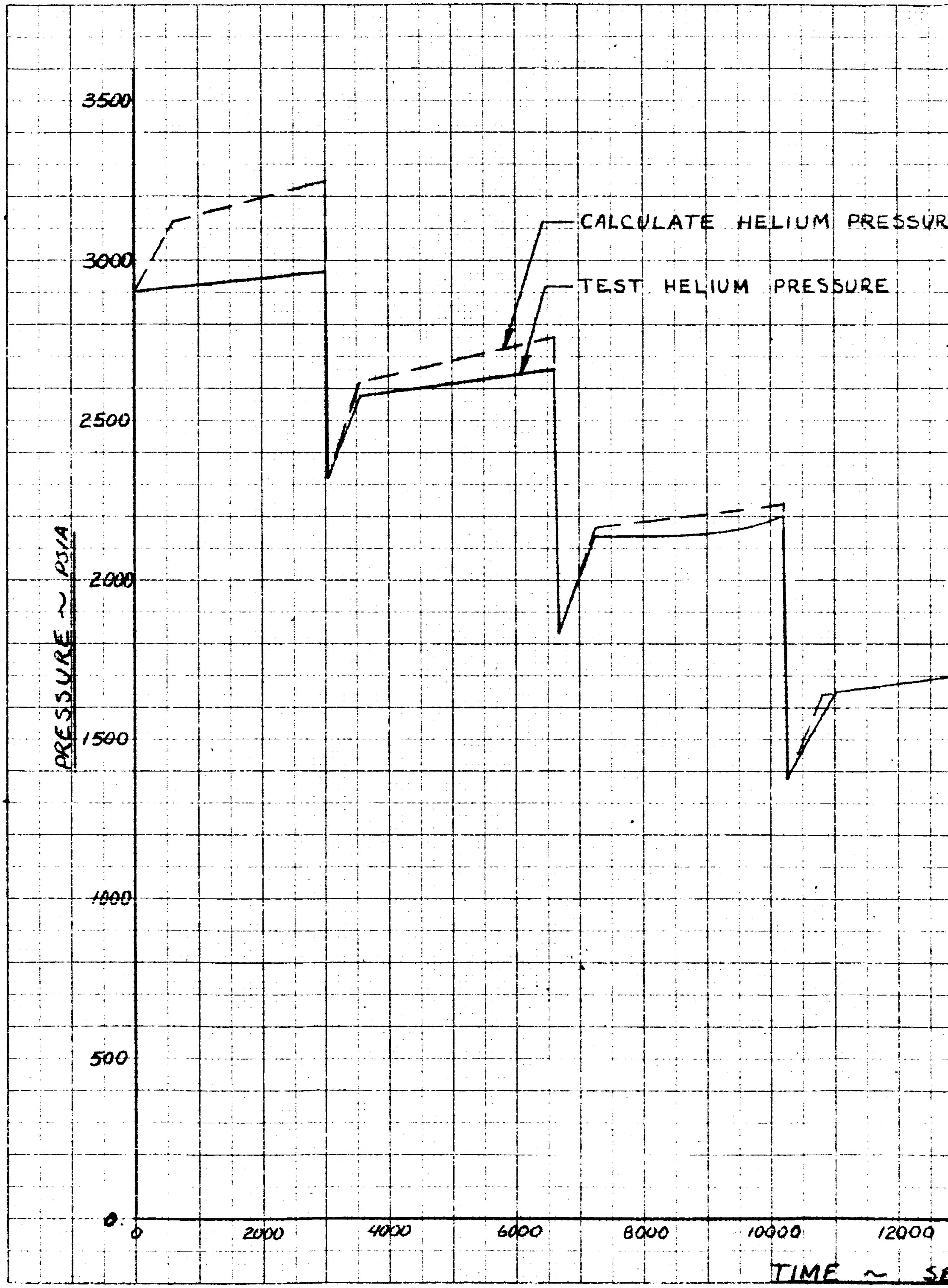


FIGURE
TEST AND CALCULATED DATA
HELIUM STORAGE TEST # 6
5-65
TEMPERATURE - °R
500
450
400
350
300
250
200
150
0
2000
4000
6000
8000
10,000
12,000
TIME

V-28-1



STRUCTURE TEMPERATURE
INSULATION TEMPERATURE
TEST WALL TEMPERATURE
CALCULATED WALL TEMPERATURE
CALCULATED HELIUM TEMPERATURE
TEST HELIUM TEMPERATURE
14,000
16,000
18,000
20,000
22,000
24,000
26,000
- SEC


CALCULATE HELIUM PRESSUR
TEST HELIUM PRESSURE
PRESSURE ~ PSIA
3500
3000
2500
2000
1500
1000
500
0
0
2000
4000
6000
8000
10000
12000
TIME ~ SE

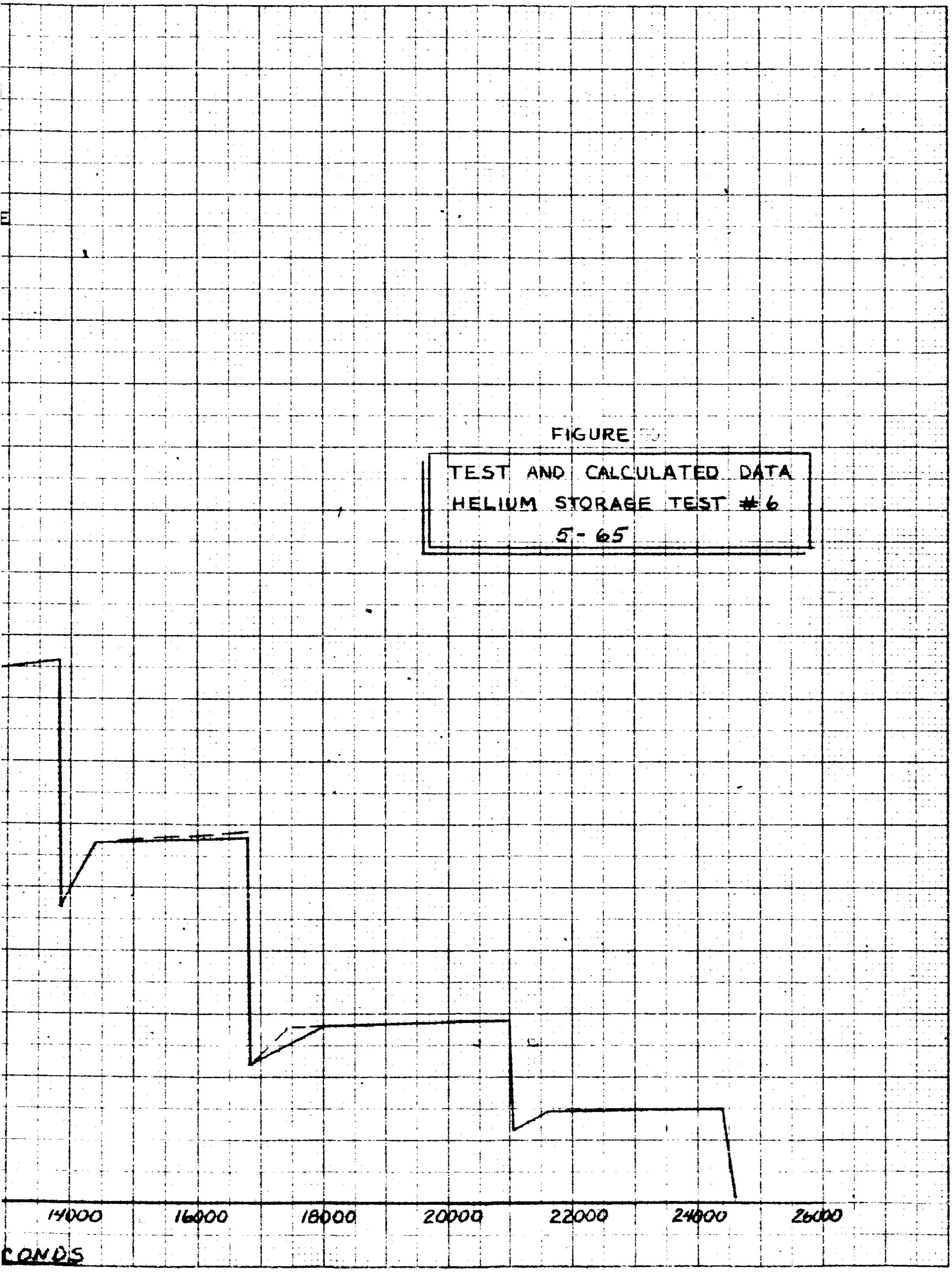


FIGURE
TEST AND CALCULATED DATA
HELIUM STORAGE TEST #6
5-65
14000
16000
18000
20000
22000
24000
26000
CONDS

helium temperature increased at a much faster rate than the test helium temperature during the short time immediatly following venting. This tendency also explains the large difference in calculated and test helium pressure during the first and second coast periods. The start of the first coast period in both tests clearly shows that the calculated helium temperature increased rapidly to within 2 degrees of the calculated wall temperature and then began the gradual temperature rise while the test helium temperature rise was gradual during the entire coast. Except for the small deviations mentioned above, the calculated helium pressures, helium temperatures, and wall temperatures compared favorably with the helium pressures, average helium temperatures, and average wall temperatures experienced during the two tests when instrumentation accuracy is considered. The accuracy of the thermocouples at the temperatures experienced during the tests was $\pm 8°F$. The accuracy of the pressure transducer was ± 50 psi. Therefore, on the basis of this evaluation and comparison, the computer program did provide a good simulation of actual conditions, within the accuracy of the measurements.

2. Propellant Feed Line Heat Exchanger Tests

Objective - The objective of this test was to determine the operating characteristics of a propellant feed line heat exchanger, similar to that of the Apollo Service Propulsion System, using ambient temperature N_2H_4/UDMH fuel to heat 160°R helium.

Test Fixture - The entire test fixture is shown schematically in Figure 51. A more detailed drawing of the actual heat exchanger is shown in Figure 52. N_2H_4/UDMH fuel was supplied to the heat

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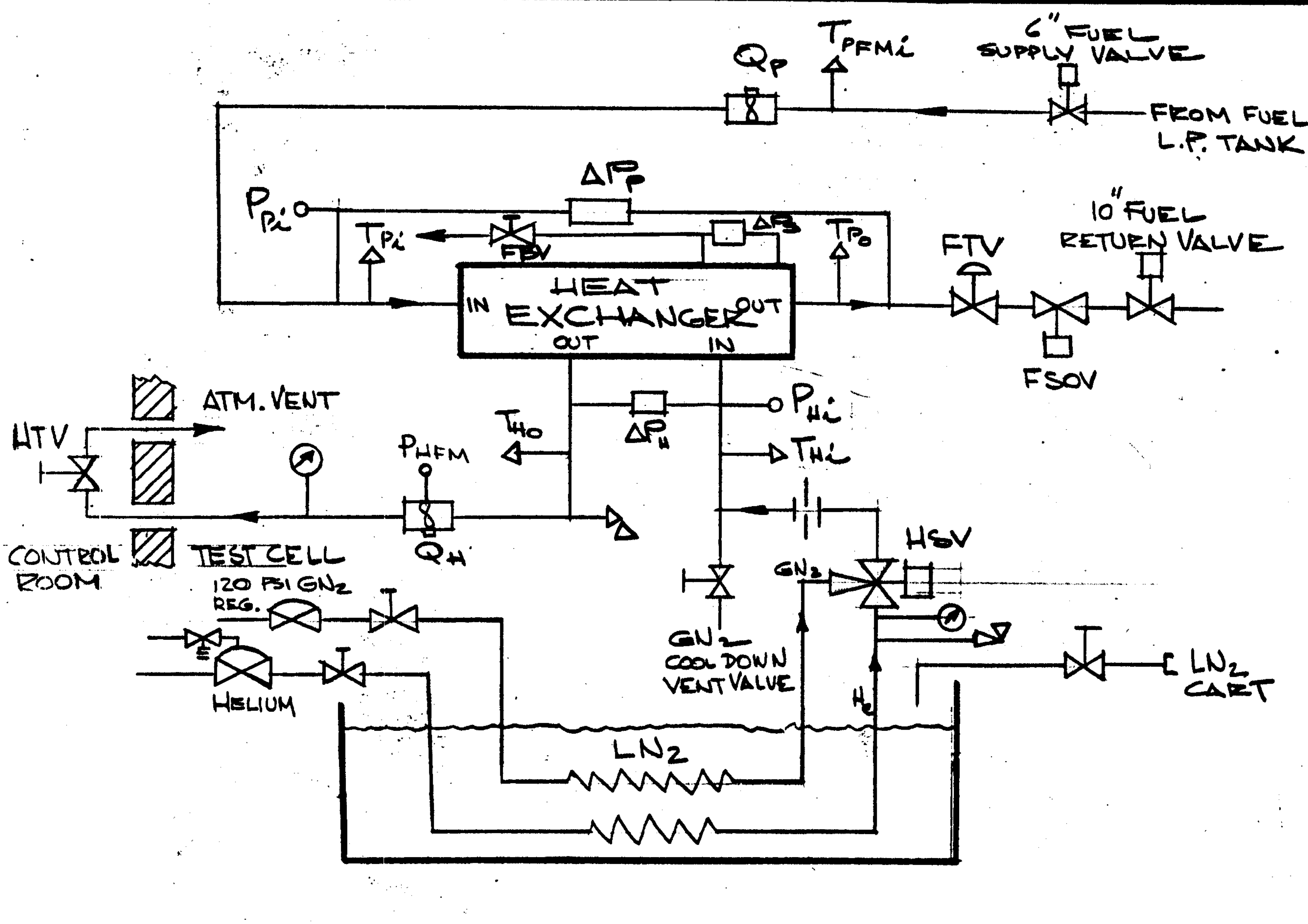
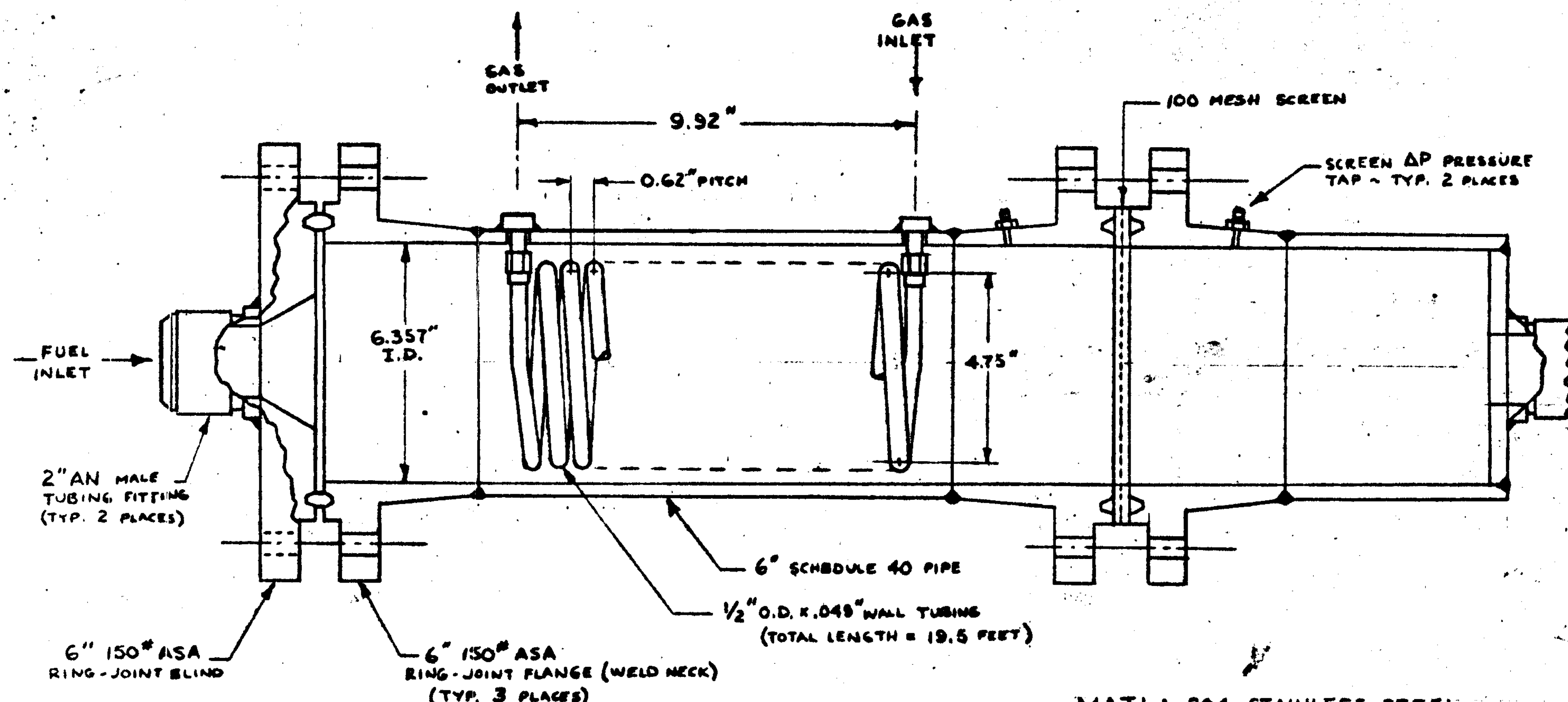
FIGURE 51 FEED LINE HEAT EXCHANGER TEST SCHEMATIC





MATL: 304 STAINLESS STEEL

FIGURE 32
FEED LINE HEAT EXCHANGER
SCALE: 1/4" = 1"

J.B.K. 6-25-65

exchanger from the facility supply tank. The fuel circuit was equipped with a remote shut-off valve (FSOV) and a remotely controlled flow throttling valve (FTV). The heat exchanger was installed in the fuel circuit so that the fuel passed through the shell of the exchanger. The gas circuit of the heat exchanger consisted of 19 feet of ½ inch diameter by .049-inch thick wall, stainless steel tubing wound in a helix having a pitch of 0.62 inches and a pitch-line radius of 2-3/8 inches. The coil length, measured from the inlet fitting to the outlet fitting centerlines, was 9.92 inches. The heat exchanger shell (6 inch, schedule 10 stainless steel pipe) was 6.357 inches I.D. with a wall thickness of 0.134 inches. The fuel outlet end of the heat exchanger was fitted with one thickness of 100-mesh stainless steel screen to serve as an ice-catcher. Static pressure bosses were provided upstream and downstream of the screen, to provide for measurement of pressure drop across the screen. The screen and pressure taps were provided to determine whether or not freezing of the fuel occurred as a result of transferring heat to the cold helium. The actual test fixture is shown in Figure 53.

Helium was supplied to the heat exchanger inlet from a liquid nitrogen soak-tank heat exchanger. The soak tank contained 248 feet of one-inch diameter by .110-inch thick wall, stainless tubing through which helium was passed at approximately 1700 psig. A small smooth approach orifice or flow nozzle was provided in the helium supply line between the soak tank and the test heat exchanger helium inlet. This flow nozzle

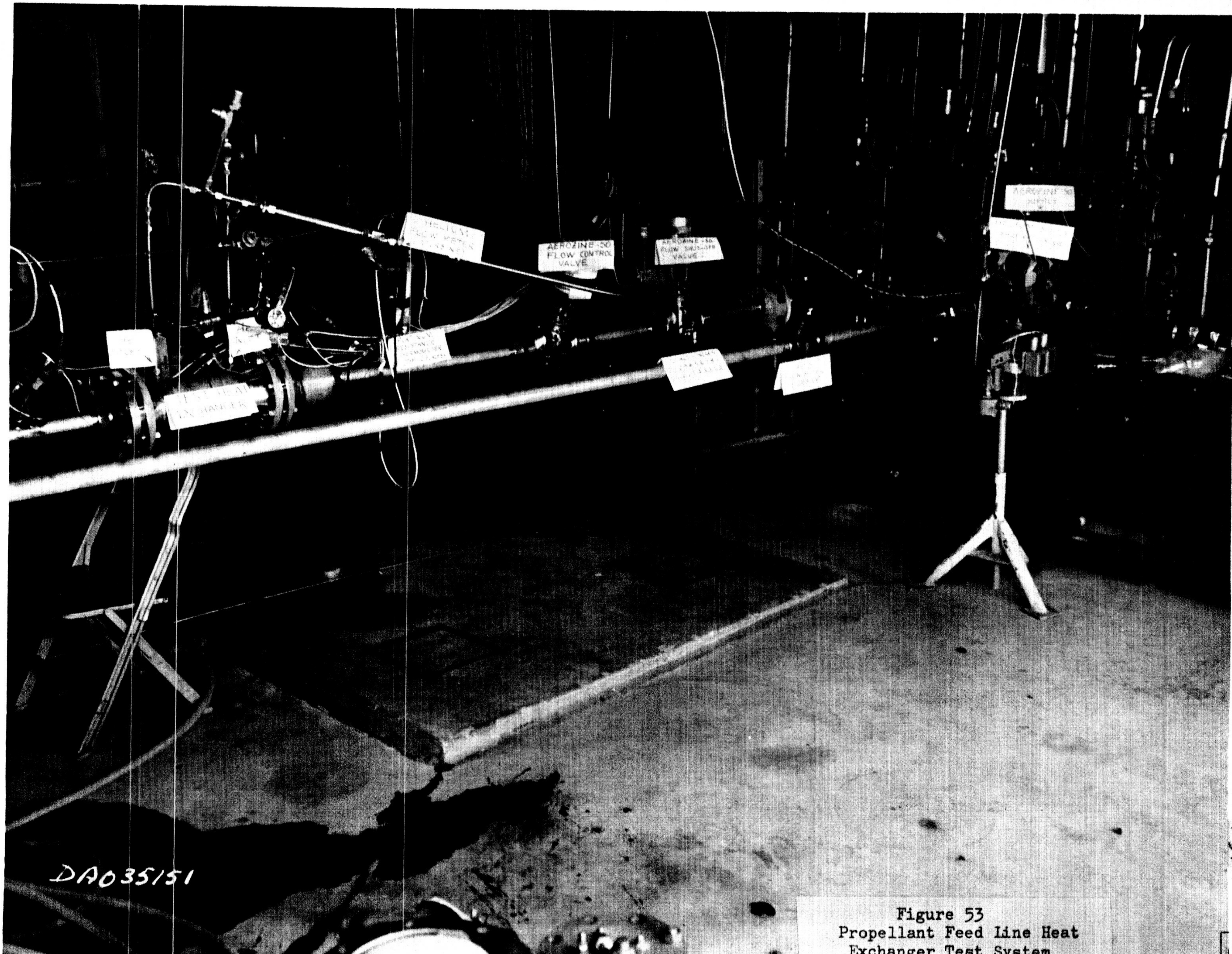


Figure 53
Propellant Feed Line Heat
Exchanger Test System

metered the helium flow to the test fixture in response to the supply pressure imposed (desired mass flow rate was .07 lbs/sec). The heat exchanger helium discharge piping was equipped with a turbine flow meter and a manual throttling or back-pressure adjusting valve, to permit adjustment of the helium outlet pressure. As a result of the use of a choked flow nozzle in the helium supply line, the heat exchanger helium discharge pressure could be adjusted over a wide rante without changing the mass flow through the system. The helium supply from the soak tank was fitted with a remotely-operated, shut-off valve to permit rapid onset and termination of helium flow through the test heat exchanger.

Procedure - The helium soak tank was filled with LN_2; cool-down occurred until steady-state conditions were obtained. The fuel circuit through the test heat exchanger was bled in to remove any trapped gas. A flow adjustment run was then conducted. At this time, the N_2H_4/UDMH fuel circuit was pressurized to approximately 90 psig and the throttling valve was adjusted to obtain a flow rate of 185 to 190 gallons per minute (approximately 23 pounds per second). With the fuel flowing at this rate, the helium supply through the soak tank was pressurized to the approximate 1700 psig pressure required to obtain a mass flow of .07 lbs/second of 160°R helium gas through the flow nozzle. The helium discharge throttling valve was then adjusted as required to obtain a pressure of 175 psia at the heat exchanger helium outlet. Adjustments to the helium supply pressure and heat exchanger discharge throttle were then made as required to verify the desired helium mass flow. The helium supply was then shut-off with the remote shut-off valve, without changing the supply pressure setting or the discharge throttle setting. The fuel supply was then shut-off by closing the facility 6" supply valve to the fixture. After the fuel flow had stopped, the test fixture remote shut-off valve (FSOV) was closed and the 6" facility supply valve was re-opened. This technique was employed in order to avoid excessive hydraulic shock on shut-down and to utilize the quick-acting FSOV valve to re-start the fuel flow. Prior to the test run, cold nitrogen gas was allowed to flow through the supply line and out the by-pass valve at the inlet to the heat exchanger, to pre-cool the relatively massive HSV 3-way helium supply valve. This GN_2 purge was not used at any time during the actual test run.

The actual test run was initiated by simultaneously opening the helium supply valve (HSV) and the fuel shut-off valve (FSOV). The system was allowed to reach steady state, as determined by the valve adjustments made during the flow adjustment period; no adjustments were made during the run. After six (6) minutes of steady-state operation (simulated sustained flight), the helium flow and fuel flow were stopped simultaneously. After a ten (10) minute hold period (simulated coasting flight), helium flow and fuel flow were initiated simultaneously for a two (2) minute period of steady-state operation. At the end of the two minute period, fuel and helium flows were stopped simultaneously for another 10 minute coast. At the end of this coast period, another 2 minute period of operation was accomplished and the run was terminated. Data recorders were run continuously at normal chart speeds during the operational periods and at slow speed during the coast periods. The pressure drop across the ice-catcher screen was monitored during operation; no excessive ice formation was detected.

Discussion - A series of three check-runs and two test runs were made, using propellant at ambient temperature and helium gas at approximately 160°R. Each of the two test runs, Runs 4 and 5, consisted of alternating periods of operation (simulated sustained flight) and shut-down (simulated coasting). Helium mass flows of .05 to .07 lbs/sec., and propellant flows of 180 to 190 GPM were employed. With these conditions, the helium temperature was raised from 160°R to approximately 520°R (within approximately 15 degrees of the propellant supply temperature). Fuel temperature drop through the heat exchanger was approximately 2°F. Satisfactory data were obtained to define heat transfer and pressure drop characteristics and to ascertain that no fuel freezing problem existed. Steady-state periods of operation of up to six (6) minutes duration were employed, and start-up and shut-down transient histories were recorded. A comparison of the experimental and analytical results is presented below.

Analysis of Results - There were five propellant feed line heat exchanger tests that utilized fuel to heat helium. The first three were check-out runs, and the last two were complete runs. The complete runs, Tests 4 and 5, were the only runs that were evaluated and compared with analytical data.

Analytical simulation was accomplished with the existing IBM 1620 gas-to-liquid heat exchanger computer program. The test data input to the program were helium inlet and outlet temperatures, fuel inlet temperature, helium flow rate, fuel flow rate, and helium inlet pressure. Also input were the physical and thermodynamic properties of helium and propellant, and the test heat exchanger configuration, i.e., tube I.D., O.D., length and wall thickness, and feed line cross sectional area. The computer program calculated the heat exchanger length, helium pressure drop, and fuel temperature drop.

The analytical simulation was conducted for the steady-state portions of Tests 4 and 5 only. For Test 4, the six-minute steady-state run is designated as Test 4-A, and the two two-minute runs are designated as Test 4-B and Test 4-C. For Test 5, the six-minute, two-minute, and two three-minute steady-state runs are designated as Tests 5-A, 5-B, 5-C, and 5-D, respectively. The results of the evaluation and comparison for Test 4 are shown in Figures 54, 55, and 56. The results for Test 5 are shown in Figures 57, 58, 59, and 60. The calculated helium pressure drop compares favorable with the measured test pressure drop for Tests 4-A, 4-B, and 4-C. The difference between the calculated and measured helium pressure drop is 1 to 3 psi out of a total helium pressure drop of 51 psi for Tests 4-A, 4-B and 4-C. The calculated drop in fuel temperature across the heat exchanger also compared favorably with the measured temperature drop for Tests 4-A, 4-B, and 4-C. The calculated heat exchanger lengths were between 29.5% to 37% higher than the length of the test heat exchanger for Tests 4-A, 4-B, and 4-C. The comparison between the calculated and test heat exchanger length, helium pressure drop, and fuel temperature

FIGURE 54

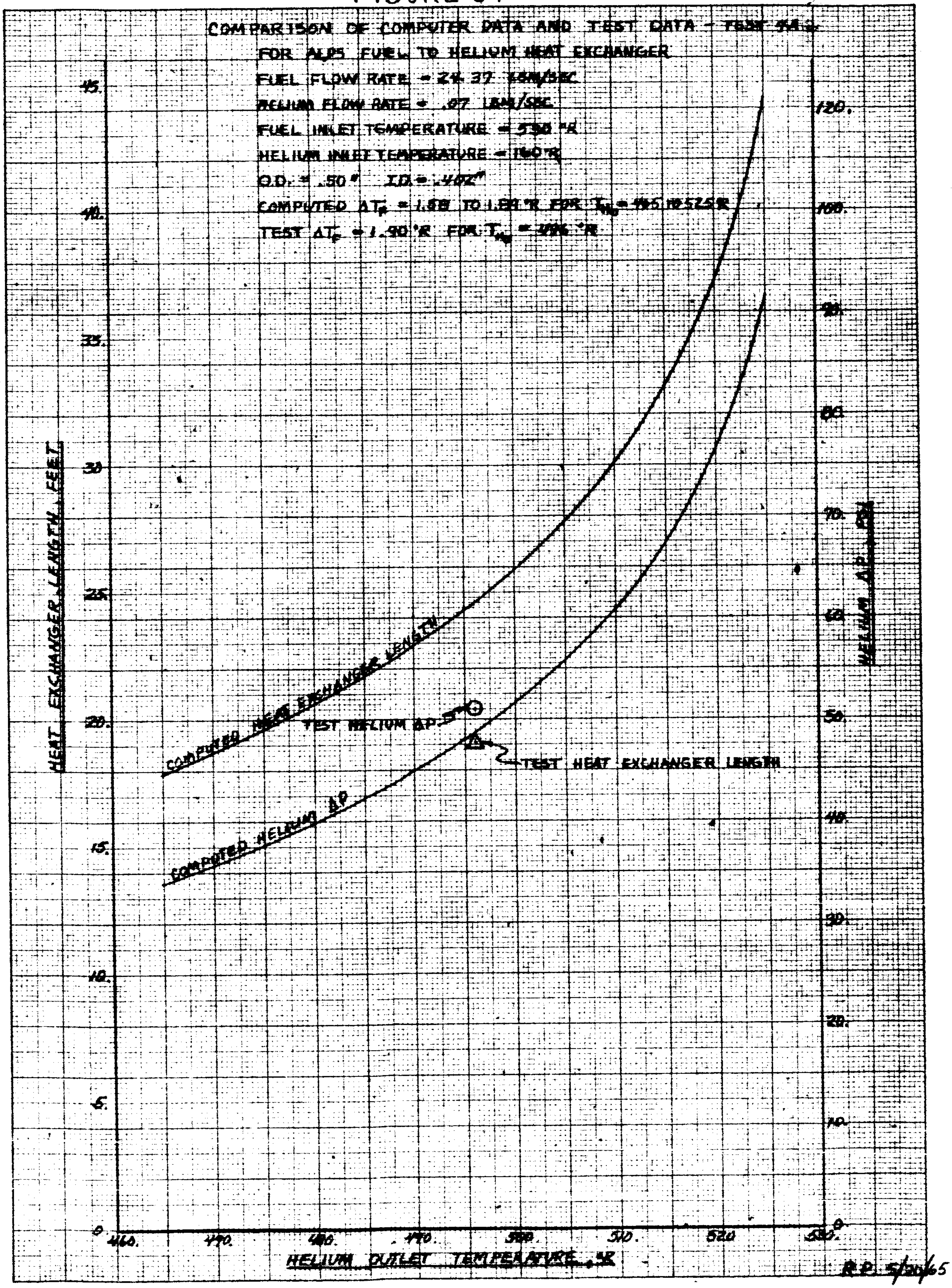
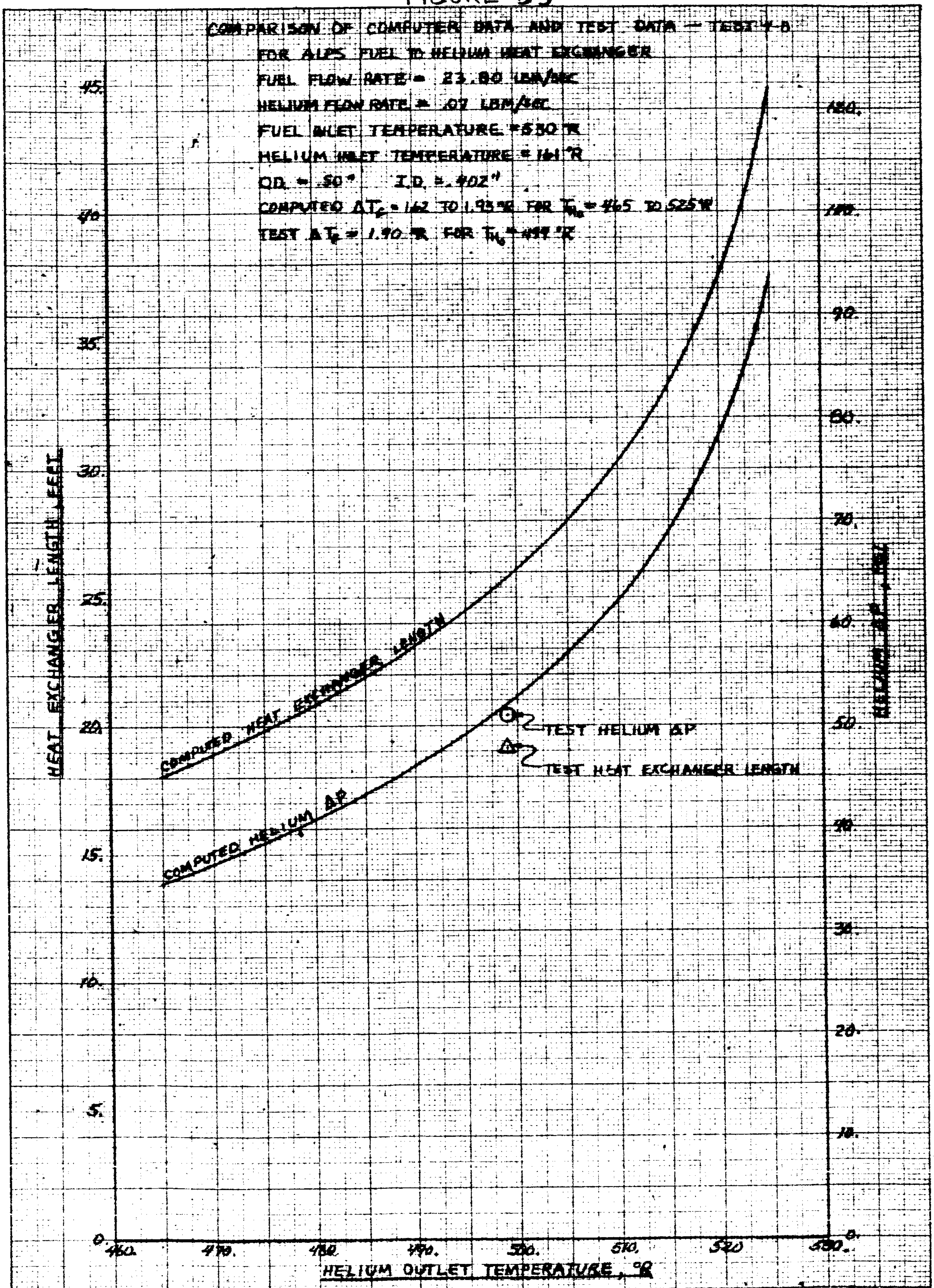

COMPARISON OF COMPUTER DATA AND TEST DATA
FOR ALPS FUEL TO HELIUM HEAT EXCHANGER
FUEL FLOW RATE = 24.37 LBM/SEC
HELIUM FLOW RATE = .07 LBM/SEC
FUEL INLET TEMPERATURE = 530 °R
HELIUM INLET TEMPERATURE = 160°R
O.D. = .50" I.D. = .402"
COMPUTED ΔT_F
TEST ΔT_F = 1.90 °R
HEAT EXCHANGER LENGTH, FEET
HELIUM ΔP, PSI
COMPUTED HEAT EXCHANGER LENGTH
TEST HELIUM ΔP
TEST HEAT EXCHANGER LENGTH
COMPUTED HELIUM ΔP
0
5.
10.
15.
20.
25.
30
35.
40.
45.
10.
20.
30.
40.
50.
60.
70.
80.
100.
120.
460.
470.
480.
490.
500.
510.
520
530.
HELIUM OUTLET TEMPERATURE, °R
R.P. 5/20/65

FIGURE 55



R.P. 5/21/65

FIGURE 56

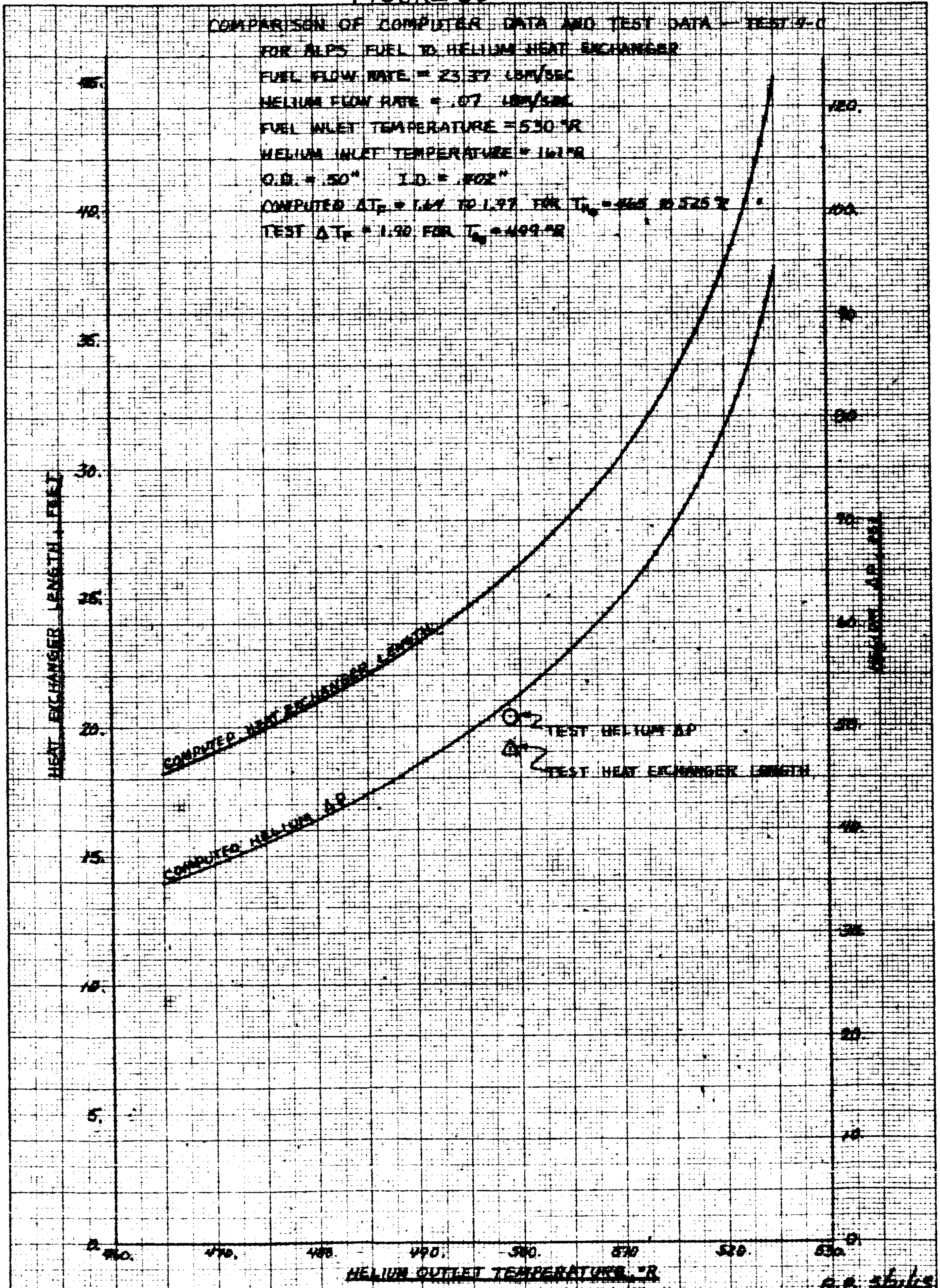

COMPARISON OF COMPUTER DATA AND TEST DATA — TEST 9-C
FOR ALPS FUEL TO HELIUM HEAT EXCHANGER
FUEL FLOW RATE = 23.37 LBM/SEC
HELIUM FLOW RATE = .07 LBM/SEC
FUEL INLET TEMPERATURE = 530 °R
HELIUM INLET TEMPERATURE = 161 °R
O.D. = .50" I.D. = .402"
COMPUTED ΔT_F = 1.64 TO 1.97 FOR T_{H_e} = 465 TO 525 °R
TEST ΔT_F = 1.90 FOR T_{H_e} = 499 °R
COMPUTED HEAT EXCHANGER LENGTH
COMPUTED HELIUM ΔP
TEST HELIUM ΔP
TEST HEAT EXCHANGER LENGTH
HEAT EXCHANGER LENGTH, FEET
HELIUM OUTLET TEMPERATURE, °R
0.
5.
10.
15.
20.
25.
30.
35.
40.
45.
460.
470.
480.
490.
500.
510
520
530.
0
10.
20.
30.
40.
50.
60.
70.
80.
90.
100.
120.
R.B. 5/21/65

FIGURE 57

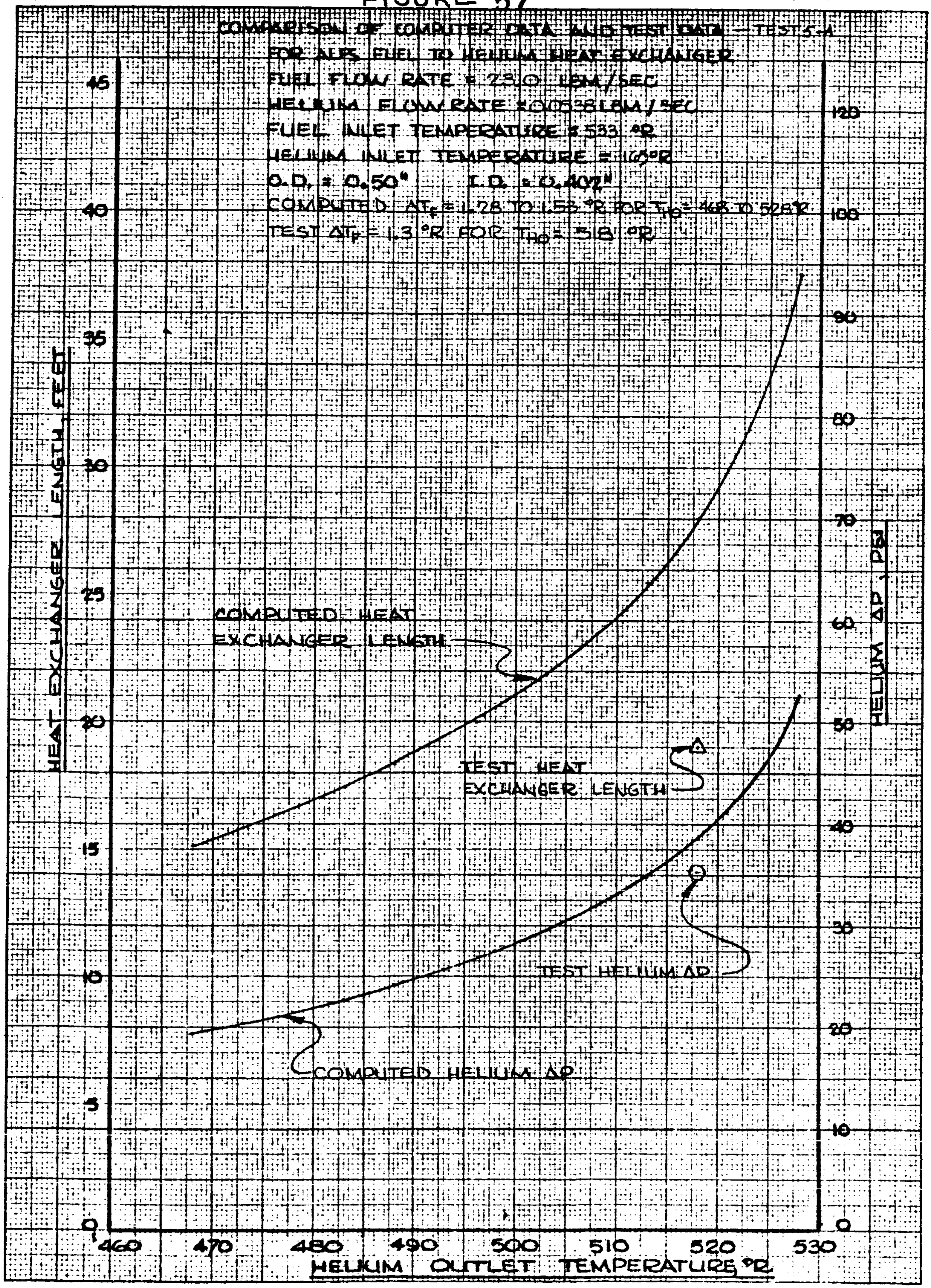

COMPARISON OF COMPUTER DATA AND TEST DATA — TEST 5-A
FOR ALPS FUEL TO HELIUM HEAT EXCHANGER
FUEL FLOW RATE = 23.0 LBM/SEC
HELIUM FLOW RATE = 0.0538 LBM/SEC
FUEL INLET TEMPERATURE = 533 °R
HELIUM INLET TEMPERATURE = 165°R
O.D. = 0.50"   I.D. = 0.402"
COMPUTED ΔT_F = 1.28 TO 1.58 °R FOR T_HO = 468 TO 528°R
TEST ΔT_F = 1.3 °R FOR T_HO = 518 °R
COMPUTED HEAT EXCHANGER LENGTH
TEST HEAT EXCHANGER LENGTH
TEST HELIUM ΔP
COMPUTED HELIUM ΔP
HEAT EXCHANGER LENGTH, FEET
HELIUM ΔP, PSI
HELIUM OUTLET TEMPERATURE °R
0
5
10
15
20
25
30
35
40
45
0
10
20
30
40
50
60
70
80
90
100
120
460
470
480
490
500
510
520
530

FIGURE 58

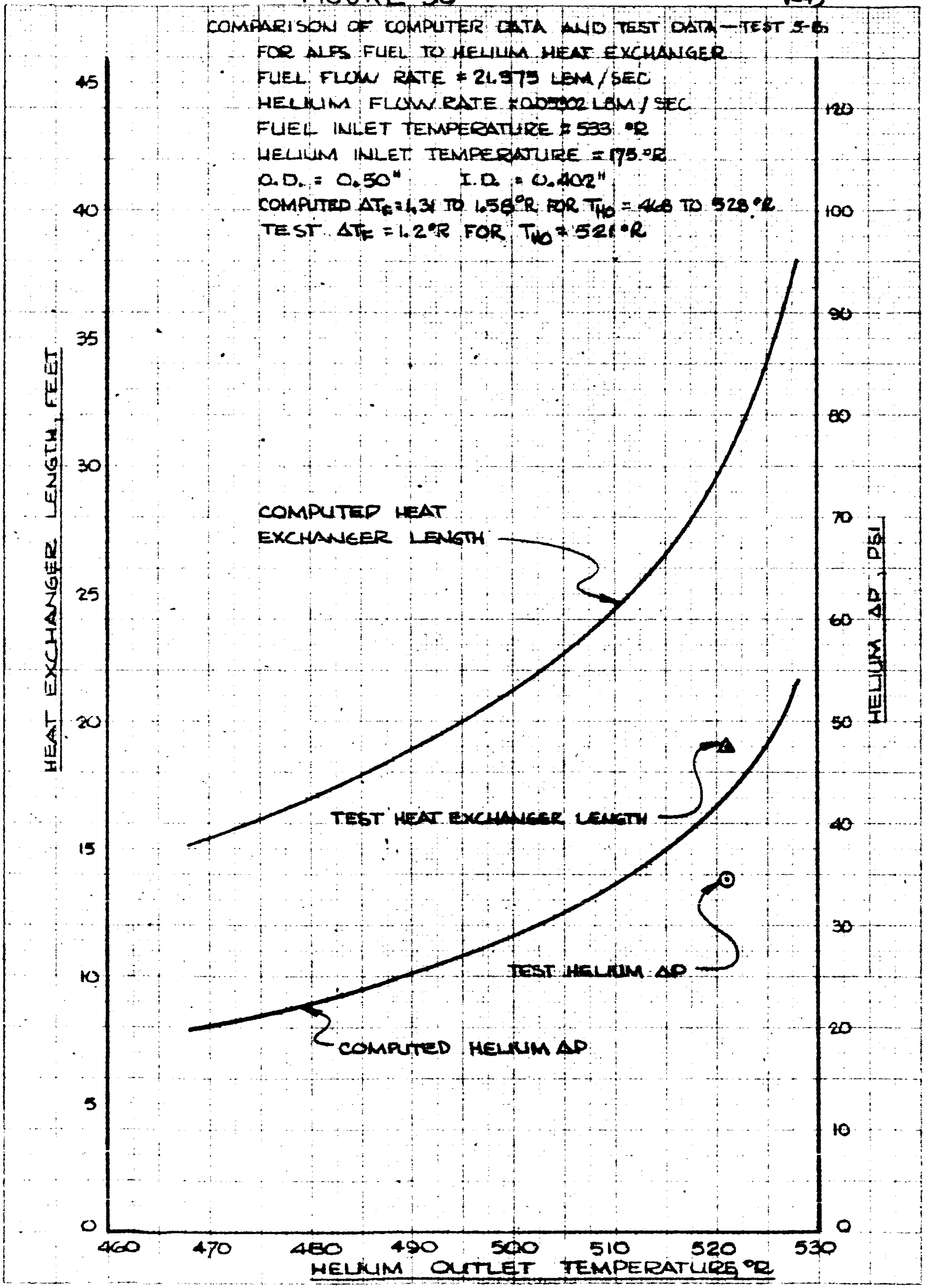

COMPARISON OF COMPUTER DATA AND TEST DATA — TEST 5-6
FOR ALPS FUEL TO HELIUM HEAT EXCHANGER
FUEL FLOW RATE = 21.975 LBM / SEC
HELIUM FLOW RATE = 0.05902 LBM / SEC
FUEL INLET TEMPERATURE = 533 °R
HELIUM INLET TEMPERATURE = 175 °R
O.D. = 0.50"   I.D. = 0.402"
COMPUTED ΔT_F = 1.31 TO 1.58 °R FOR T_{HO} = 468 TO 528 °R
TEST ΔT_F = 1.2 °R FOR T_{HO} = 521 °R
COMPUTED HEAT EXCHANGER LENGTH
TEST HEAT EXCHANGER LENGTH
TEST HELIUM ΔP
COMPUTED HELIUM ΔP
HEAT EXCHANGER LENGTH, FEET
HELIUM ΔP, PSI
HELIUM OUTLET TEMPERATURE °R
0
5
10
15
20
25
30
35
40
45
0
10
20
30
40
50
60
70
80
90
100
120
460
470
480
490
500
510
520
530

FIGURE 59

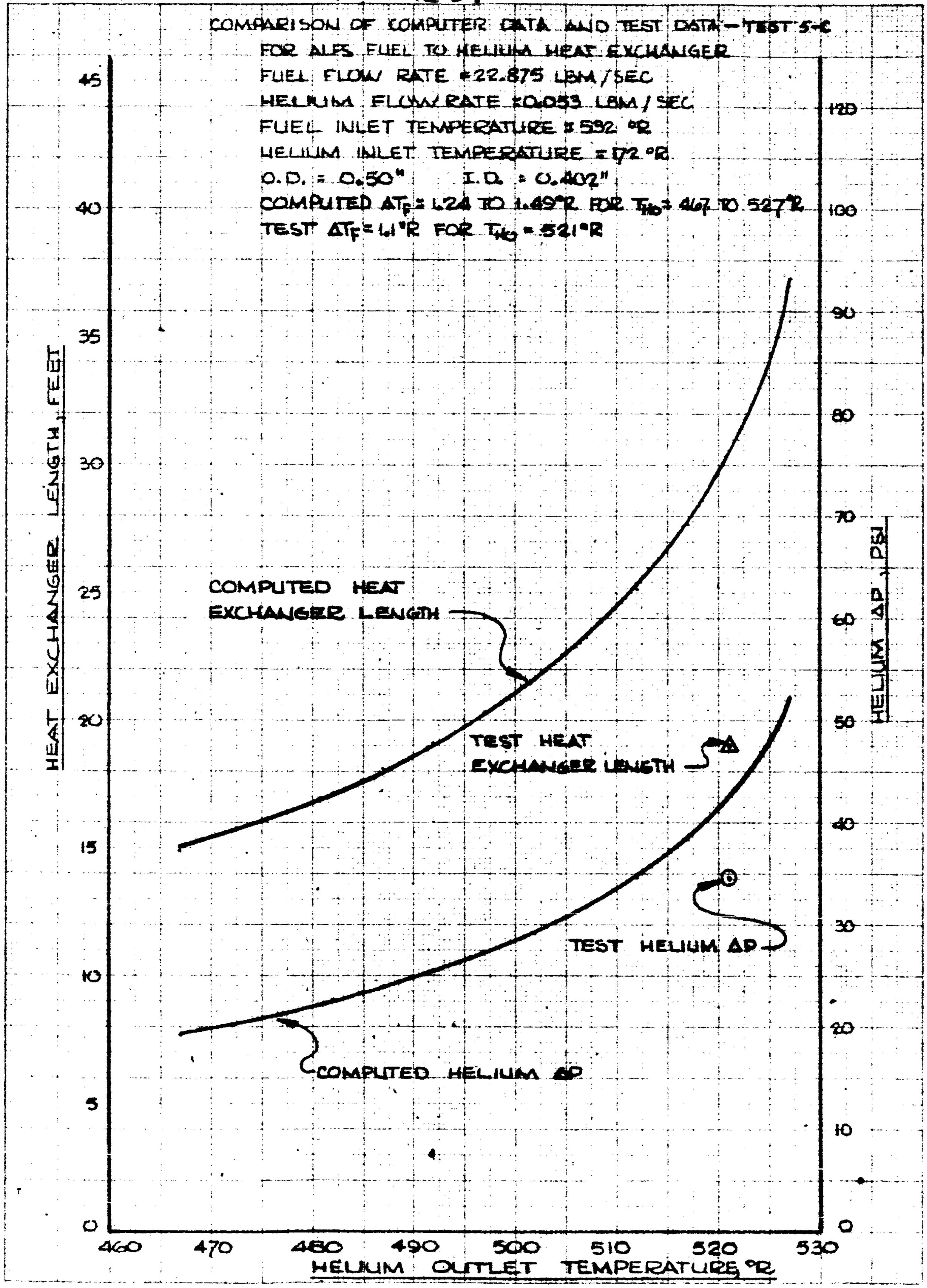
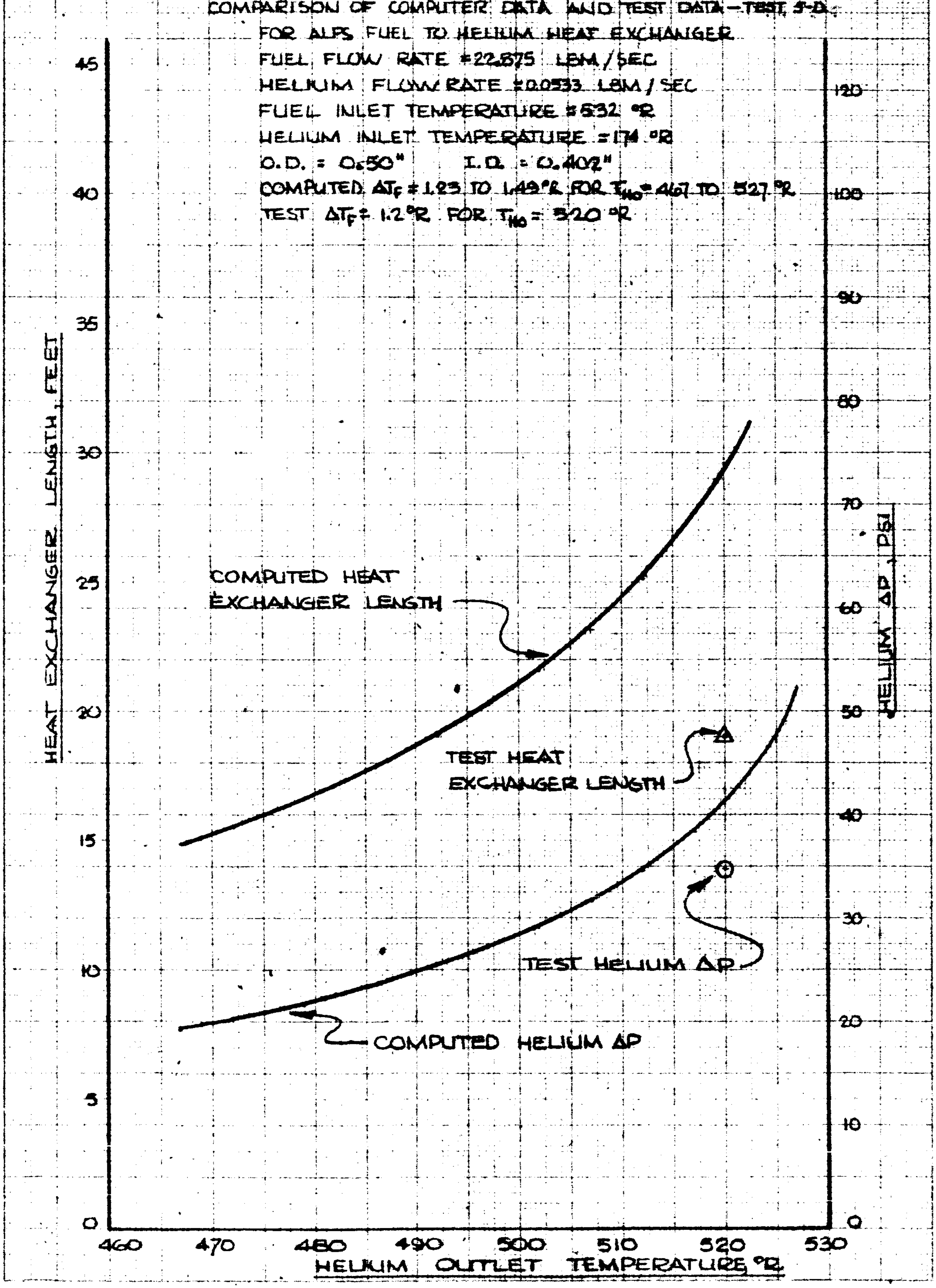

COMPARISON OF COMPUTER DATA AND TEST DATA — TEST 5-E
FOR ALPS FUEL TO HELIUM HEAT EXCHANGER
FUEL FLOW RATE = 22.875 LBM/SEC
HELIUM FLOW RATE = 0.053 LBM/SEC
FUEL INLET TEMPERATURE = 532 °R
HELIUM INLET TEMPERATURE = 172 °R
O.D. = 0.50"   I.D. = 0.402"
COMPUTED ΔT_F = 1.24 TO 1.49°R FOR T_{HO} = 467 TO 527°R
TEST ΔT_F = 1.1°R FOR T_{HO} = 521°R
COMPUTED HEAT EXCHANGER LENGTH
TEST HEAT EXCHANGER LENGTH
TEST HELIUM ΔP
COMPUTED HELIUM ΔP
HEAT EXCHANGER LENGTH, FEET
HELIUM ΔP, PSI
HELIUM OUTLET TEMPERATURE, °R
0
5
10
15
20
25
30
35
40
45
460
470
480
490
500
510
520
530
0
10
20
30
40
50
60
70
80
90
100
120

FIGURE 60


COMPARISON OF COMPUTER DATA AND TEST DATA — TEST 5-D
FOR ALPS FUEL TO HELIUM HEAT EXCHANGER
FUEL FLOW RATE = 22.875 LBM/SEC
HELIUM FLOW RATE = 0.0533 LBM/SEC
FUEL INLET TEMPERATURE = 532 °R
HELIUM INLET TEMPERATURE = 174 °R
O.D. = 0.50"   I.D. = 0.402"
COMPUTED ΔT_F = 1.25 TO 1.49 °R FOR T_{HO} = 467 TO 527 °R
TEST ΔT_F = 1.2 °R FOR T_{HO} = 520 °R
HEAT EXCHANGER LENGTH, FEET
HELIUM ΔP, PSI
HELIUM OUTLET TEMPERATURE °R
COMPUTED HEAT EXCHANGER LENGTH
TEST HEAT EXCHANGER LENGTH
TEST HELIUM ΔP
COMPUTED HELIUM ΔP
0
5
10
15
20
25
30
35
40
45
460
470
480
490
500
510
520
530
10
20
30
40
50
60
70
80
90
100
120

drop was not as close for Test 5 as was obtained for Test 4. The calculated heat exchanger lengths were 47.5% to 58% higher than the length of the test heat exchanger for Tests 5-A, 5-B, 5-C and 5-D. The calculated helium pressure drop was from 2.5 to 8.0 psi higher than the observed pressure drop. The calculated fuel temperature drop compared favorably (within 0.4°R) with the test fuel temperature drop.

Propellant pressure drop across the heat exchanger was measured at 4.0 to 5.7 psia at the design flow rate. A calculation of the entrance and exit losses was performed using the Darcy equation ($P = K \rho V^2/2gc$). These losses account for 3.3 psi of the observed mean pressure drop of 4.8 psi. The pressure drop occurring in the propellant flowing through the core of the heat exchanger was not calculated, because no empirical flow resistance coefficient could be found for the test unit configuration.

On the basis of this evaluation and comparison, the analytical model will give quite conservative values for heat exchanger sizes. The primary source of disagreement between the calculated and actual heat exchanger length was the equation employed to predict the heat transfer coefficient across the inside gas film. The computer program employed an equation for a straight tube with moderate temperature rise while in actuality the tube was a helical coil with high gas temperature rise. Use of the correction for a helical coil increases the inside film coefficient by about 30% to 40% for the actual exchanger dimensions and results in a decrease in the predicted lengths. This effect is being incorporated into the computer program along with other refinements, i.e., temperature at which properties are evaluated and more precise exponents. Much closer agreement between predicted and measured values should result; however, the analytical model will still be maintained on the slightly conservative side.

3. Pulse-Mode Pressurization System Tests

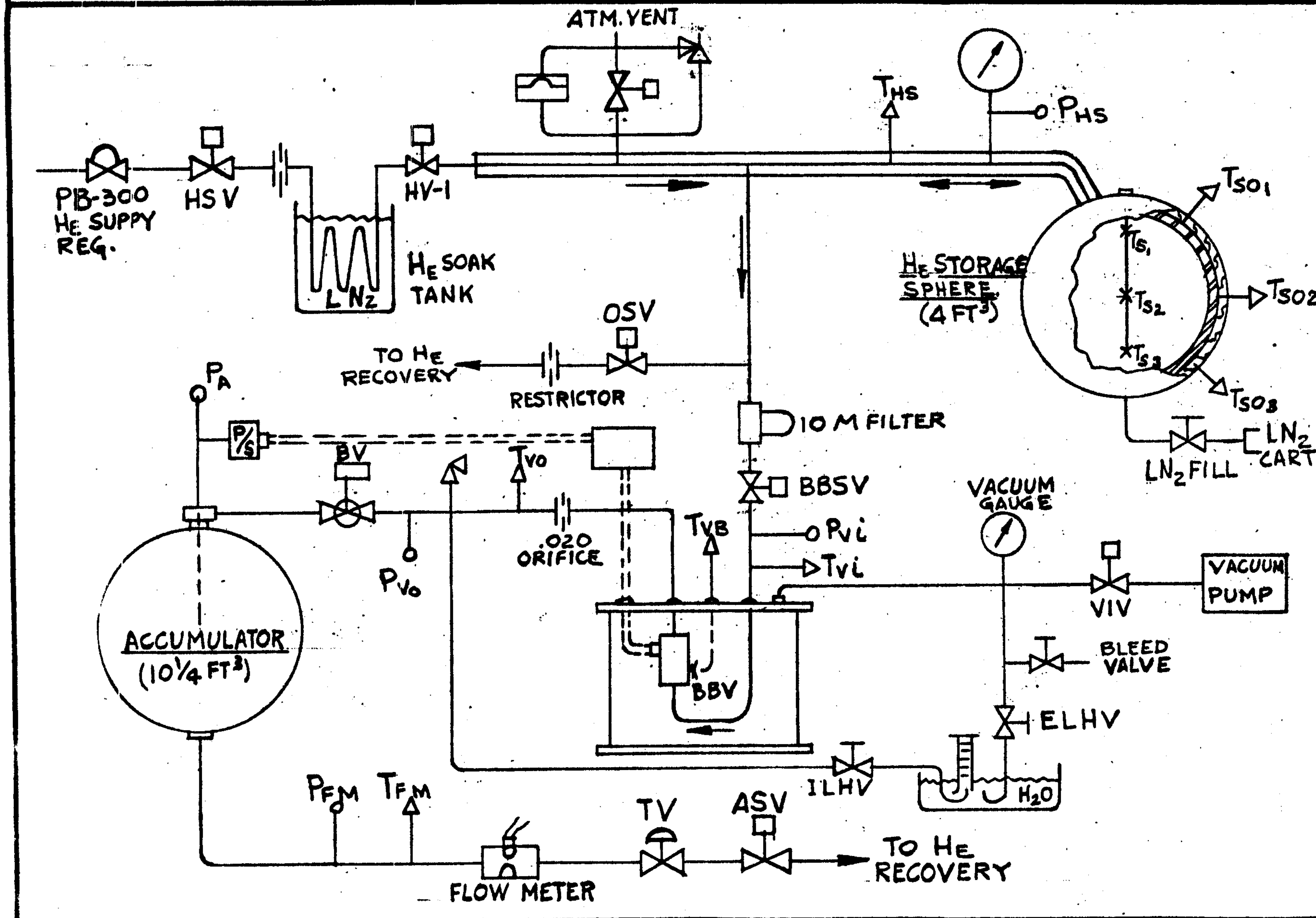
Objective - The objective of this test was to obtain empirical information on a pulse-mode pressure control subsystem which might be incorporated in a propellant tank pressurization system. It was desired that the subsystem control the ullage pressure in a simulated propellant tank within a narrow band when supplied 140°R to 160°R helium from a stored source. Further, the subsystem had to function properly at helium flow rates from zero (coasting flight) to values associated with full-thrust sustained flight.

Test Fixture - The test fixture was composed of the 4-cubic foot, insulated storage sphere (same unit used in the helium storage test), a Sterer 3/4" solenoid-operated shut-off valve (P/N 28370), a Hydra Electric 155 psig pressure switch and a 10¼-cubic foot accumulator sphere (simulated minimum propellant tank-top ullage). A .020-inch diameter, sharp-edged orifice was installed downstream of the Sterer shut-off valve to reduce its flow capacity to the desired range. A remotely-operated throttling valve was installed in the discharge line from the accumulator sphere to permit adjustment of helium mass flow rate to the desired values of .06 to .10 lbs/sec. Remotely controlled shut-off valves were installed at appropriate points in the system to permit fast-response starting and stopping of the helium flow from the accumulator and to isolate the Sterer valve for purposes of leak checking. A schematic diagram of the system is shown in Figure 61; the actual installation is shown in Figure 62.

The power supply to the Sterer valve was controlled by the pressure switch, with provisions for manual over-ride on the control console. The pressure switch was wired to send 28 VDC power to the valve whenever the accumulator pressure dropped below the 155 ± 1 psig set-point, thus calling for the valve to open. The Sterer valve was mounted in a small vacuum chamber having uninsulated walls. A vacuum pump was provided to evacuate the chamber to approximately 0.2 psia.

<u>Procedure</u> - The insulated storage sphere was loaded through the LN_2-helium heat exchanger until the desired initial helium storage conditions of 2000 psig and 160°R were attained. During this loading phase, the system was pressurized up to the BBSV shut-off valve located upstream of the Sterer valve (BBV). The BBV valve was isolated in this manner because the valve required a minimum inlet pressure of 150 psig in order to effect a shut-off condition, and blow-through would prevent build-up of pressure in the storage sphere. During the early part of the loading, at approximately 300 psig, with the BBV valve solenoid de-energized (over-ride from console switch), the BBSV isolation valve was opened to pressurize the inlet of the BBV valve. A small amount of gas, sufficient to pressurize the accumulator to about 5 psig, flowed through the BBV valve before it closed.

FIGURE 61 SOLENOID VALVE TEST SYTEM



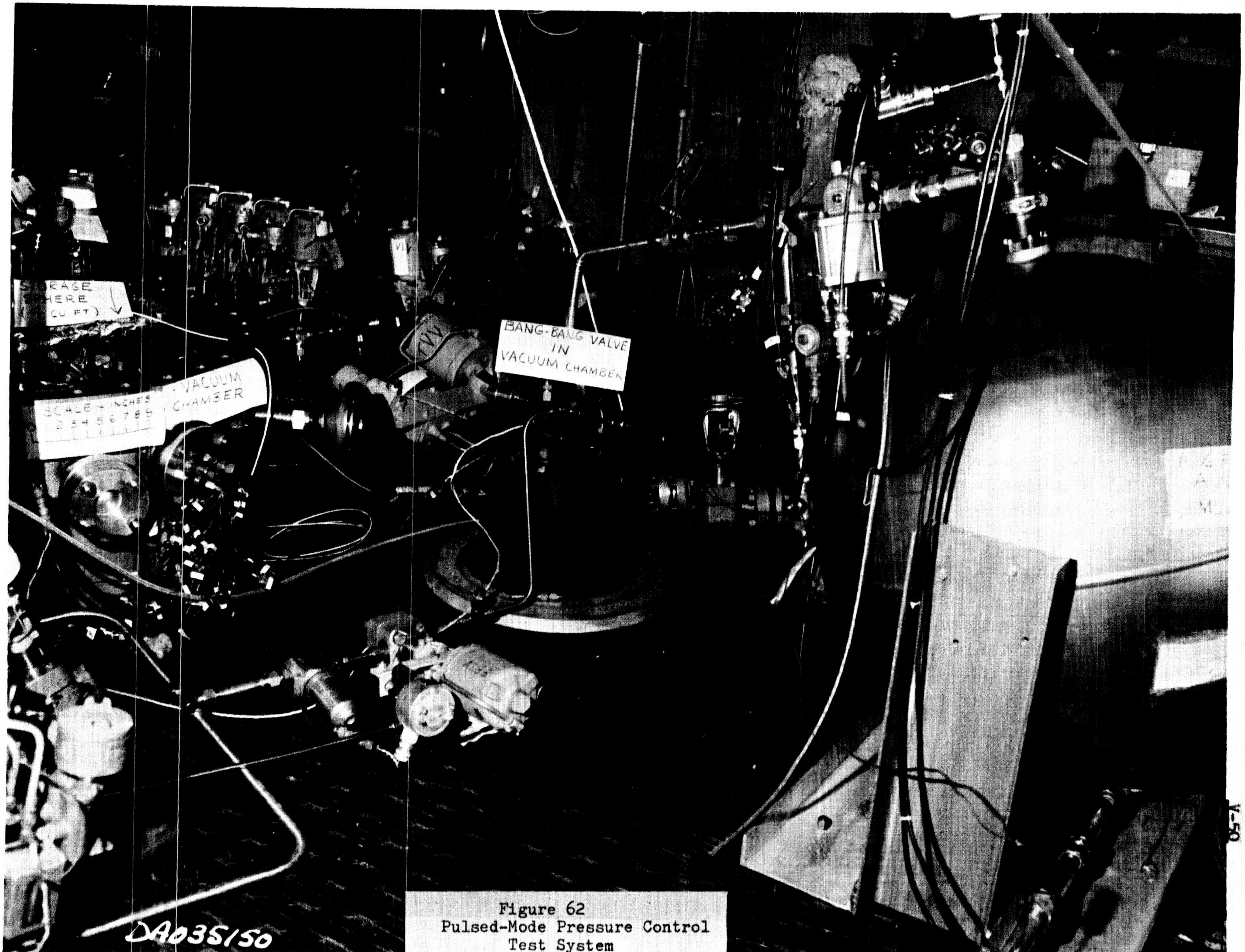


Figure 62
Pulsed-Mode Pressure Control
Test System

With the supply system pressurized to 2000 psig up to the inlet of the BBV valve, and with the accumulator outflow shut-off valve (ASV) closed, the run was initiated by transferring control of the BBV valve to the pressure switch. The pulse-mode subsystem then pressurized the accumulator to a lock-up condition of 155 $\pm$ 1 psig. At this point, the option was exercised to either reload the storage sphere (to obtain a longer run duration) or to proceed without reloading. The accumulator outlet throttle valve was then set to the position required to obtain the desired helium mass outflow. The pulse-mode operation was initiated by opening the outflow line shut-off valve (ASV). The pulse-mode subsystem maintained the accumulator pressure (P_A) at the nominal 155 psig level as the stored gas supply pressure decayed. The run was either permitted to progress continuously until the supply pressure had decayed to approximately 400 psig, or the run was interrupted periodically by closing the ASV outflow shut-off valve, causing the pulse-mode subsystem to bring the accumulator to a lock-up condition for a simulated coast period. During the coast period, at approximately 1000 psig supply pressure, the BBV valve was checked for internal and external leakage. All instrumentation functions were recorded continuously during the run. The one recorder on which the more significant functions (P_{vi}, P_{vo}, P_A, etc) were recorded was run at maximum speed (100 mm/sec.) for short periods in order to permit an accurate determination of the response characteristics of the pulse-mode subsystem.

Discussion - A series of three runs were made with the stored helium gas source at an initial condition of 160°R and 2000 psia (maximum working pressure allowed for the pilot-operated shut-off valve used). Each run included an initial pressurization of the simulated tank-top ullage to a lock-up condition, followed by either a sustained pressurization run (full duration burn) or an interrupted run (burning and coasting).

Operation of the pulse-mode subsystem was satisfactory during all test runs, and acceptable data was obtained on the dynamic characteristics of the subsystem and its components at helium mass flows up to .07 lbs/sec. The results of these tests are discussed below.

Analysis of Results - Determination of the existence of any combination of conditions under which the solenoid valve would not be suitable for use in the Apollo Service Propulsion System (SPS) was desired. Primary items that would cause rejection were insufficient response, particularly at minimum propellant tank ullage, excessive leakage, stickiness, jamming, or any other non-reliable type actuation.

Response of the valve was determined to be approximately 50 milliseconds on opening and 40 milliseconds on closing when valve temperature was -30°F and helium inlet pressure was 1000 psig. These responses are representative of actuations at various conditions for all the tests. At constant valve temperature, opening response time increases with increasing pressure by about 0.5 to 2 milliseconds per 100 psi pressure rise. At constant operating pressure, the opening response time increases with increasing temperature by approximately 5 to 20 milliseconds per 100°F temperature rise. Maximum opening response time was 65 milliseconds at 1900 psig inlet pressure and 9°F valve body temperature. Minimum opening response time was 45 milliseconds at 750 psig inlet pressure and -95°F valve body temperature. The same minimum opening response time was obtained at 570 psig inlet pressure at -44°F valve body temperature. Closing response time as a function of inlet pressure and valve temperature could not be determined from the data. Maximum closing response time was 50 milliseconds at 1900 psig inlet pressure and 9°F valve body temperature. Minimum closing response time was 35 milliseconds at each of the following conditions:

Inlet Pressure		Valve Body Temperature
1400 psig	and	+43°F;
570 psig	and	-44°F;
1500 psig	and	-38°F.

Typical test data recorder traces are presented in Martin CR-64-82 (Issue 8) "Monthly Progress Report," June, 1965.

Pressure overshoot at the simulated minimum propellant tank ullage was maximum when inlet pressure was maximum and amounted to 4 psig. The pressure switch setting at 70°F was 156 psig with contact breaking with increasing accumulator pressure. Maximum accumulator pressure was 160 psig after a pressure rise rate of 80 psi/second. During outflow, accumulator pressure control was 154$\pm$3 psig at the start; this decreased to 153.5$\pm$1.5 psig as valve inlet pressure decreased from 2000 psig to 400 psig.

Leakage was determined before and after subsystem testing by positive displacement of water. Prior to testing no internal or external leakage was obtained with the valve inlet pressurized to 1500 psig and valve body temperature successively reduced to -50°F, -125°F and -170°F. Valve body temperature was reduced by flowing cold helium gas through the valve. Following subsystem testing, the valve was again tested for leakage after being chilled to -320°F in liquid nitrogen. Leakage was greater than 60 scc/sec at 500 psig, 1000 psig and 1500 psig. It was decided that the valve had not seated properly due to being closed without sufficient inlet pressure, i.e., greater than 150 psig. This had been done at the completion of subsystem tests when the valve was closed following venting of tank pressure. Another leak test was run after the valve was cycled once with adequate inlet pressure. With this proper seating, no internal or external leakage was detected with the valve body at -320°F and the inlet pressurized at 500 to 1500 psig.

Cyclic rate of the valve varied from 1.4 cps to 1.2 cps from start to finish of a typical test. The amount of time the valve was open during each complete cycle varied from 13% at the start with 1900 psig valve inlet pressure to 45% at the end with 440 psig valve inlet pressure; volumetric flow rate was 0.5 actual cubic feet per second (ACFS). At 1.0 actual cubic feet per second and 1500 psig inlet pressure, the cyclic rate was 2.0 cps with the valve open 32% of each cycle. As inlet pressure decreased at the same flow rate, the frequency dropped to 1.5 cps with the valve open 68% of each cycle.

The only evidence of pressure oscillations was in valve inlet pressure at solenoid valve closure. These oscillations were approximately ± 70 psig at 19 cps for 1900 psig inlet pressure and ± 50 psig at 15 cps for 700 psig inlet pressure. The oscillations were damped out in about 0.6 second. Solenoid valve actuation was uniform and repeatable at all times. No evidence of jamming, stickiness or other non-reliable type operation was evident during any of the tests.

The performance of the valve was satisfactory even though it was oversized. A valve specifically designed for this application would produce less pressure overshoot when pressurizing tanks at minimum ullage with maximum helium supply pressure.

4. Gas Generator/Propellant Feed Line Gas Cooler Tests

Objective - The objective of this test was to determine the operating characteristics of a pressurization subsystem comprised of a hydrazine decomposition chamber (hot gas generator) supplying pressurant gases and a propellant feed line gas cooler.

Test Fixture - The test fixture used in this test was a modification of the fixture used for the feed line heat exchanger test. The heat exchanger (gas cooler) was the same unit and the N_2H_4/UDMH propellant circuit was unchanged. The gas generation and flow system consisted of: a hydrazine supply tank having a capacity of 1.2 cubic feet (75 lbs. of hydrazine); a 1/2-inch, solenoid-operated shut-off valve in the hydrazine line at the gas generator inlet; a Rocket Research Company, Model RB2-100, Reaction Chamber (hot gas generator of the hydrazine decomposition type) mounted on the gas inlet side of the gas cooler; and a gas discharge throttling system composed of a fixed exit orifice in parallel with a remotely operated throttle valve.

The gas throttling system was developed when it became apparent that the gas generator was subject to destructive detonation if allowed to discharge into an inadvertantly closed system. The provision of a fixed orifice in the gas discharge system precluded the possibility of operating the gas generator with a closed exit. A remotely-controlled gas sample collector system was provided to permit collection of a gas sample during the run. The system is shown in Figures 63 and 64.

In order to promote trouble-free operation of the gas generator, a low-flowrate, nitrogen gas purge system was connected into the gas generator hydrazine supply line between the hydrazine admission valve and the injector inlet. Valve control circuits were interconnected to provide a purge whenever the hydrazine admission valve was closed. In addition, a water spray was provided on the injector head to lower its temperature and reduce the possibility of obtaining a detonation.

<u>Procedure</u> - Detonations in the gas generator, encountered during Runs 1 and 2, necessitated revisions in the system. Prior to making Runs 3 and 4, the hydrazine flow orifice was calibrated with water. The gas exhaust throttling system, consisting of the fixed orifice and throttling valve in parallel, was flow calibrated with gaseous nitrogen to obtain the approximate settings required for given mass flows of gas having a molecular weight of 12. Vendor information on gas generator injector pressure drop characteristics were used to predict the approximate hydrazine supply pressure required.

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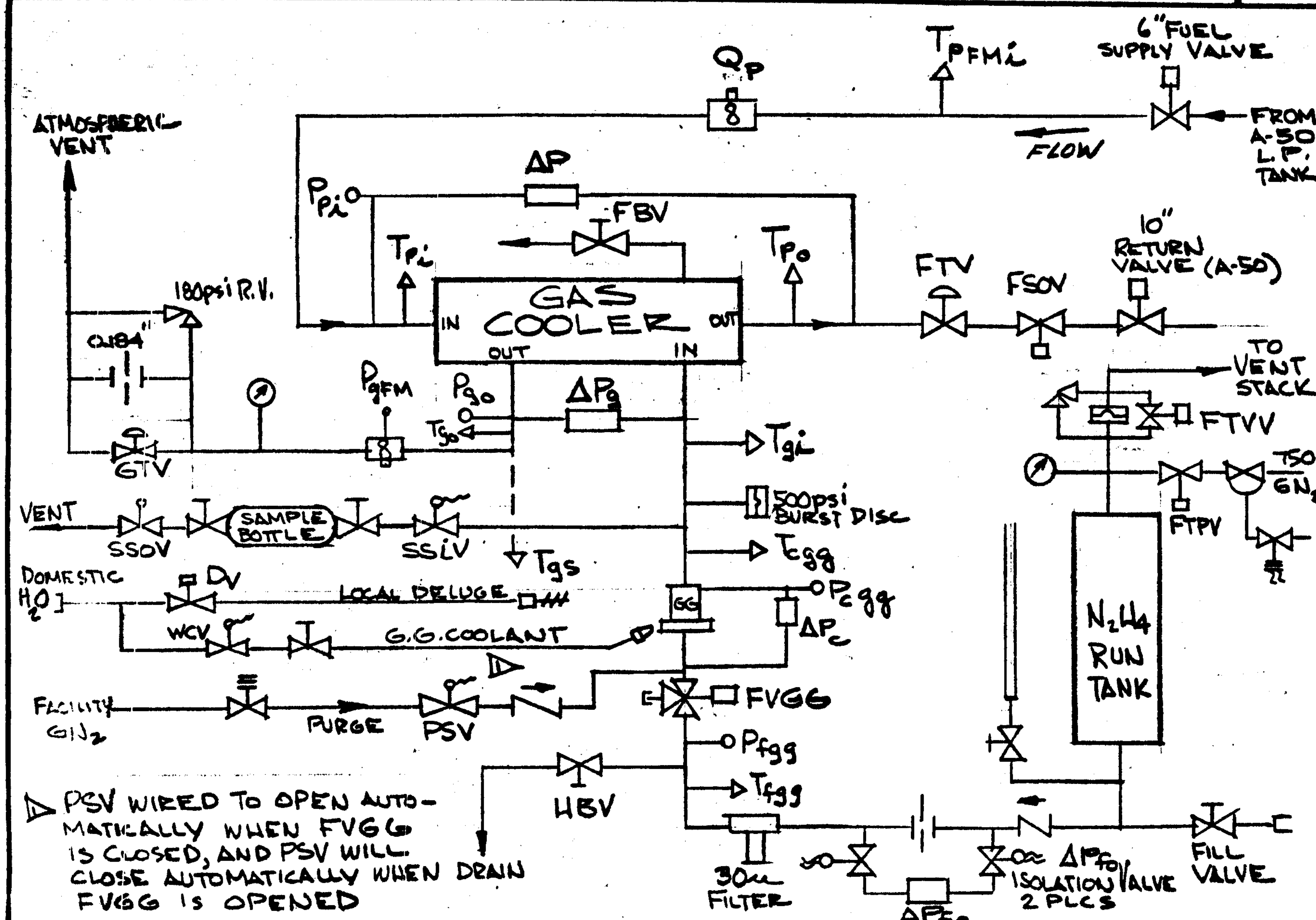
FIGURE 63 FEED LINE GAS COOLER TEST SCHEMATIC



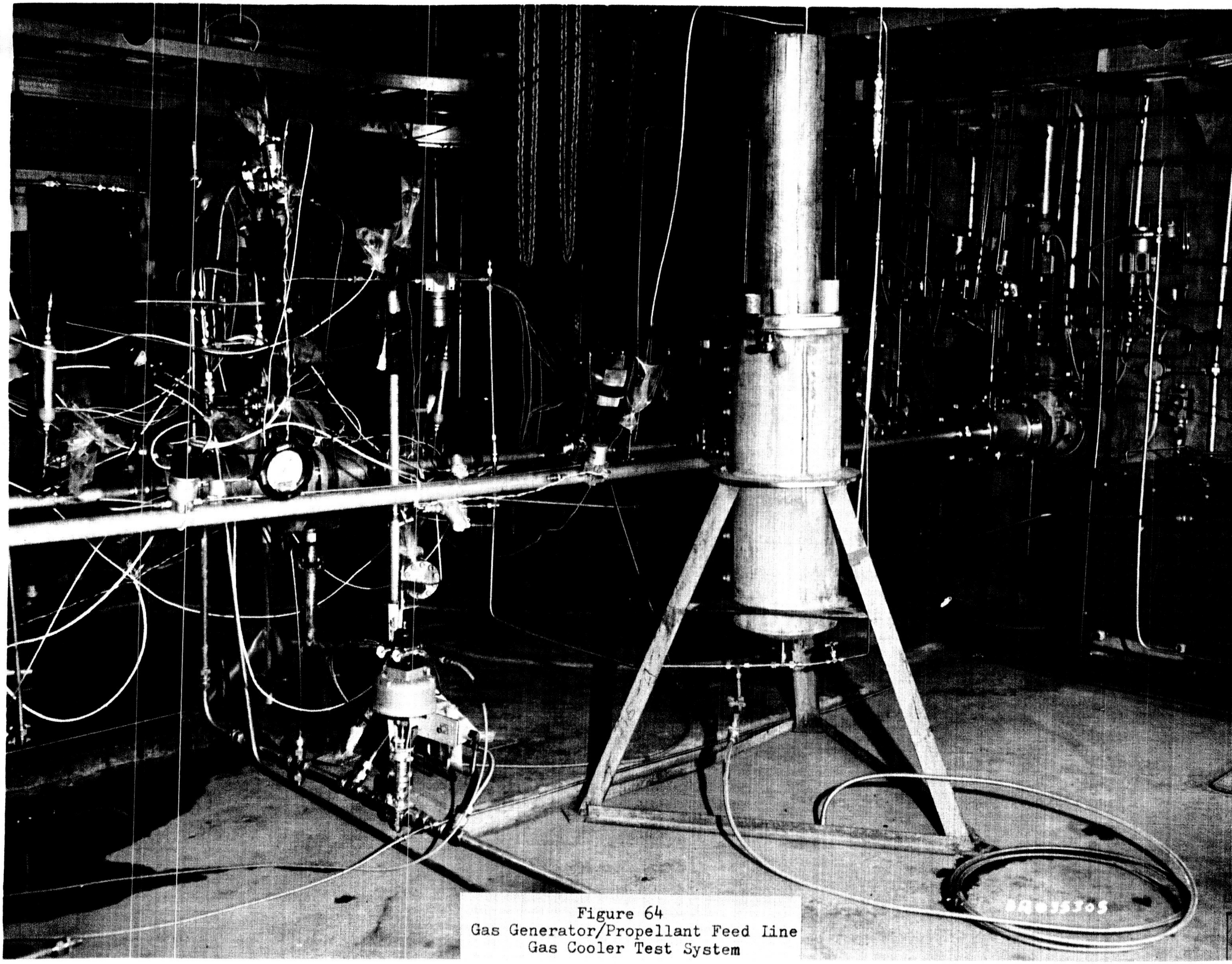


Figure 64
Gas Generator/Propellant Feed Line
Gas Cooler Test System

The first part of each test run was used for flow adjustments. This was accomplished by flowing propellant through the fuel circuit and adjusting the flow throttle (FTV) and supply tank pressure as required to obtain the design propellant flow rate of 185 to 190 gallons per minute. The gas generator hydrazine supply tank was then pressurized to the value estimated for the hydrazine mass flow desired, and the gas outlet throttle valve (GTV) was positioned to a setting calculated to produce a back-pressure of 175 psia at the cooler gas outlet with the predicted gas mass flowrate. With the geometry so set, and with the propellant flowing at the design flowrate, the gas generator was fired. Adjustments were then made to the hydrazine supply pressure and/or the gas throttle position to obtain the desired mass flowrate with the gas cooler outlet pressure (P_{go}) at 163 psig (175 psia). The gas generator and the Aerozine-50 systems were then shut down after recording pertinent set-point data and the hydrazine and Aerozine-50 supply tanks were reloaded.

The test run was started by initiating propellant flow and firing the gas generator simultaneously, using the previously established valve settings. No adjustments were made during the run. The system was allowed to run for approximately six (6) minutes. During the latter part of the run, the sampling system was closed to trap a sample of gas. At the end of the 6-minute "burn", the fuel flow and the gas generator were shut off simultaneously to simulate the start of a coast period. The surface temperature of the gas cooler tube was monitored on shut-down to detect any excessive temperature rise (above 450°F).

After a simulated coast period of 10 minutes, during which the low-rate, GN_2 purge flowed through the gas cooler circuit, a 2-minute burn was made in the same manner as the preceding 6-minute burn. Following another 10-minute coast period, a final 2-minute burn was made. After approximately one minute of the final 2-minute burn had elapsed, a short period of simulated pulse-mode operation was performed by closing and opening the gas generator hydrazine supply valve at frequencies of from 1 cycle per second to 0.25 cycle per second. The propellant flow was continuous during pulse-mode operation of the gas generator. Following completion of the test run, the gas sample bottle isolation hand-valves were closed, and the sample bottle was removed from the system.

Discussion - After resolving a series of operational problems with the gas generator, which culminated in an injector re-design, two successful full-duration runs were completed. The two runs, Runs 3 and 4, were made at gas mass flow rates of approximately .05 lbs/sec. and .13 lbs/sec, respectively. Each run consisted of a simulated 6-minute burn, 10-minute coast, 2-minute burn, 10-minute coast, and a final 2-minute burn. During the final 2-minute burn, the system was run as a manually-controlled, pulse-mode system for five to ten seconds (during pulse-mode operation, the N_2H_4/UDMH fuel flow was maintained constant).

The gas cooler was effective in cooling the gas generator exhaust products from approximately 1900°R to 540°R. Simultaneous termination of hot gas and A-50 flows were accomplished without encountering hazardous high temperatures within the propellant line. Adequate data was obtained to define the characteristics of the system; however, due to an unidentified malfunction of the gas sampling system, no gas samples were obtained. Inferential determination of the gas molecular weight was possible through use of the hydrazine input mass flow rate and the gas state conditions and volumetric flow rate at the gas cooler outlet flowmeter.

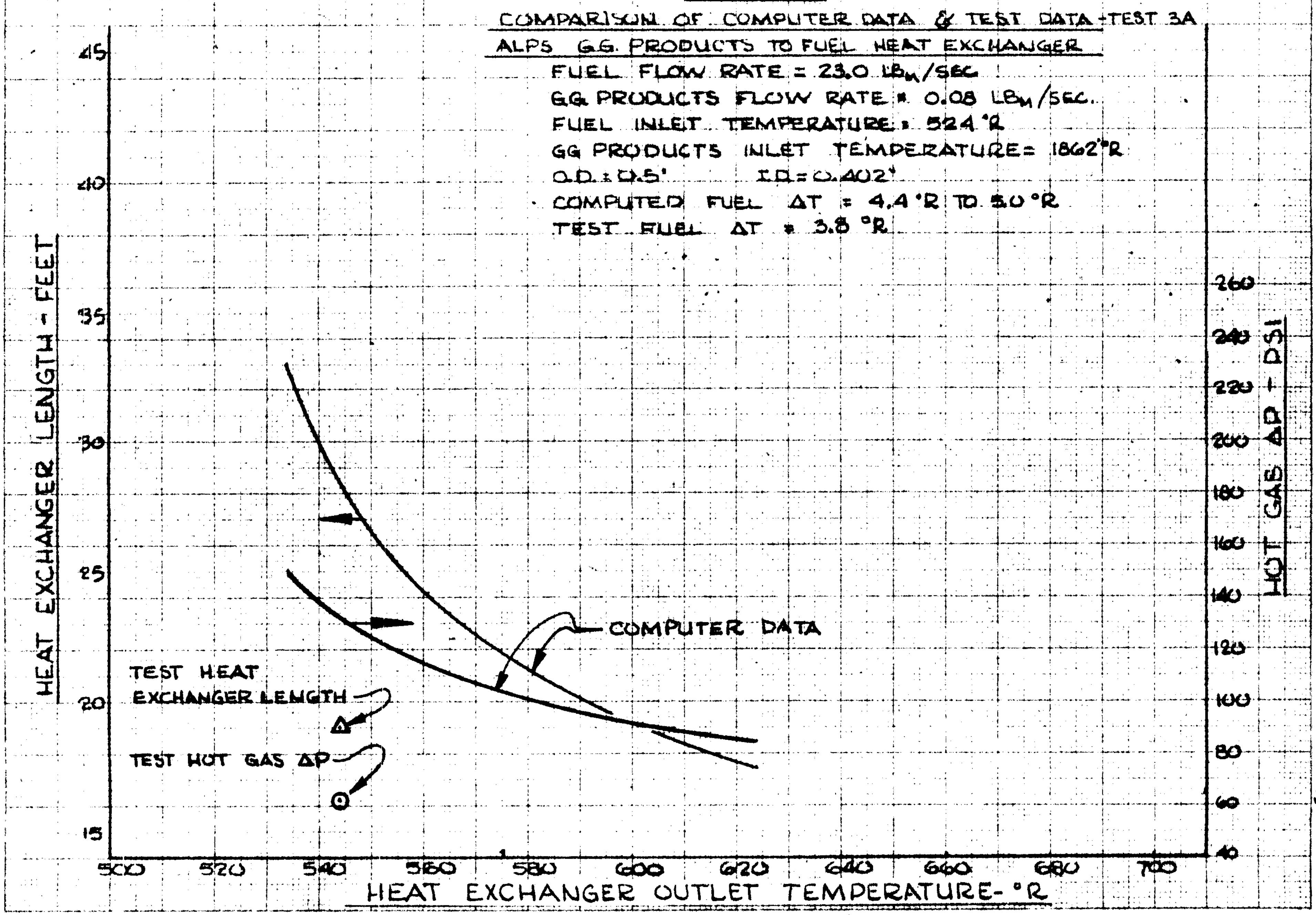
Analysis of Results - There were four hot gas to fuel feed line heat exchanger tests attempted. Of these four, the first two were not successful. The last two, Tests 3 and 4, were successful. Analytical simulation was conducted for the steady-state portions of Tests 3 and 4 only.

The analytical simulation was accomplished with the same gas-to-liquid heat exchanger computer program that was used for the helium-to-fuel heat exchanger simulation. The same type input parameters were required; output data was also the same. Values for the input parameters were taken from test data representative of the steady-state operation during Test 3 and 4. "Burns" during Test 3 consisted of a six-minute run, a two-minute run, and a two-minute pulse-mode run. Test 4 was identical to Test 3 except a higher gas flow rate was used. The pulse-mode operation for both tests was evaluated, but an analytical simulation was not attempted; pulse-mode operation of the gas generator is discussed later in this section. The six-minute and two-minute steady state runs for Test 3 are referred to as Tests 3-A and 3-B, respectively. The six-minute and two-minute steady-state runs from Test 4 are similarly referred to as Tests 4-A and 4-B, respectively.

The results of the evaluation and comparison of Tests 3-A and 3-B are shown in Figures 65 and 66. The results of Tests 4-A and 4-B are shown in Figures 67 and 68. These results are similar for all four tests. The results show that the computer data is higher than the test data for the three parameters (heat exchanger length, hot gas pressure drop, and fuel temperature rise) that were compared. The percentage difference between the computer data and test data is much greater than that observed in the previous comparisons, i.e., the comparisons on the helium-to-fuel heat exchanger. Two reasons exist for this increase between predicted and actual values. The first is due to the previously discussed need for improving the computer program equations for calculating heat transfer film coefficients (presently being accomplished) and the second is due to the unknown chemical composition of the gas generation products. Since the gas analysis from the gas generator tests were unsatisfactory, the chemical composition that was used in the analytical simulation was 59% hydrogen, 32% nitrogen and 9% ammonia (% by mole). This composition was estimated for the gas generator exit temperature range experienced during the tests. A successful

FIGURE 65

COMPARISON OF COMPUTER DATA & TEST DATA - TEST 3A
ALPS G.G. PRODUCTS TO FUEL HEAT EXCHANGER



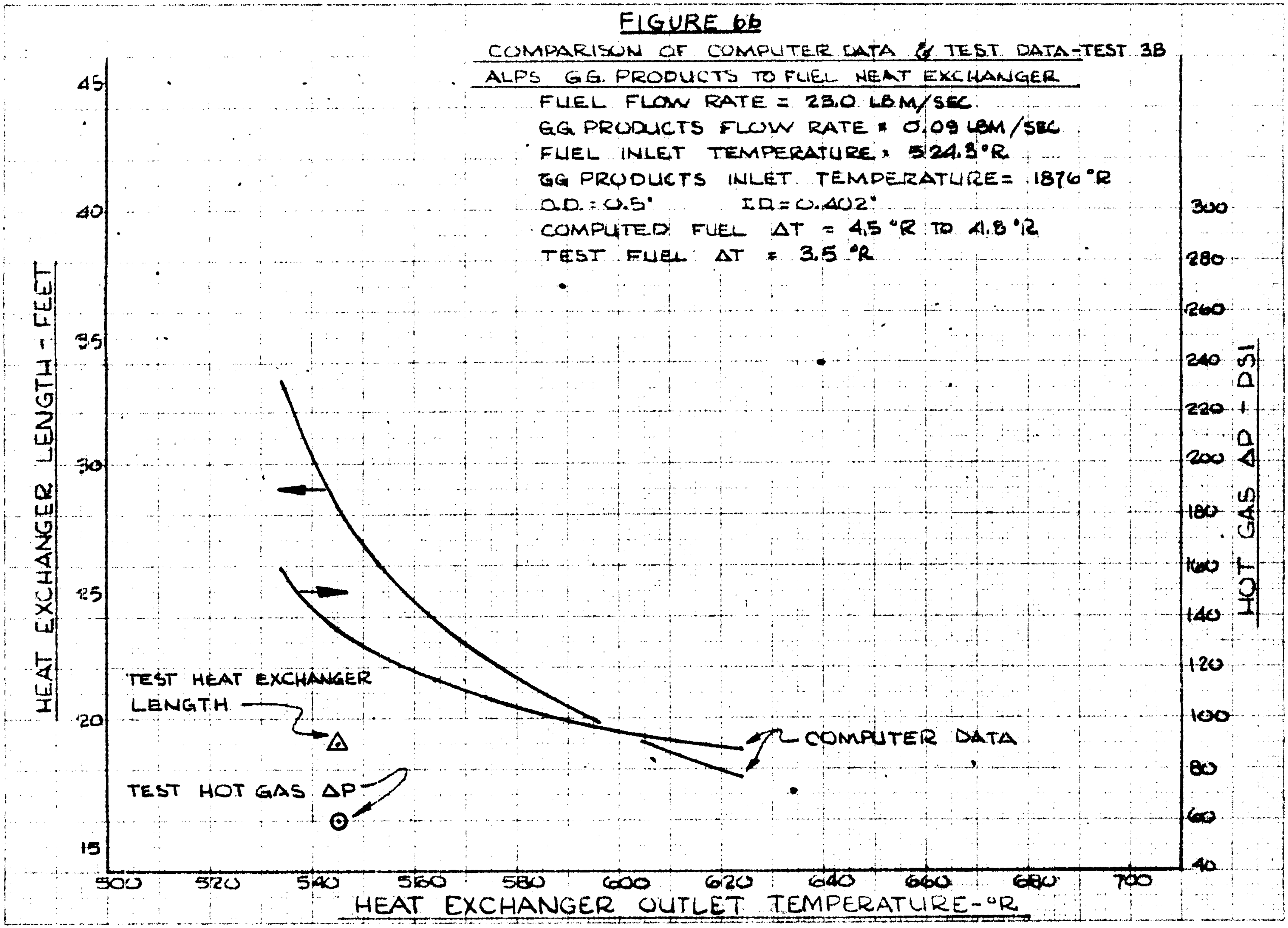


FIGURE 66
COMPARISON OF COMPUTER DATA & TEST DATA-TEST 38
ALPS G.G. PRODUCTS TO FUEL HEAT EXCHANGER
FUEL FLOW RATE = 23.0 LBM/SEC
G.G. PRODUCTS FLOW RATE = 0.09 LBM/SEC
FUEL INLET TEMPERATURE = 524.3°R
GG PRODUCTS INLET TEMPERATURE = 1876°R
O.D. = 0.5" I.D. = 0.402"
COMPUTED FUEL ΔT = 4.5°R TO 4.8°R
TEST FUEL ΔT = 3.5°R
TEST HEAT EXCHANGER LENGTH
TEST HOT GAS ΔP
COMPUTER DATA
HEAT EXCHANGER LENGTH - FEET
HOT GAS ΔP - PSI
HEAT EXCHANGER OUTLET TEMPERATURE - °R
15
20
25
30
35
40
45
40
60
80
100
120
140
160
180
200
220
240
260
280
300
500
520
540
560
580
600
620
640
660
680
700

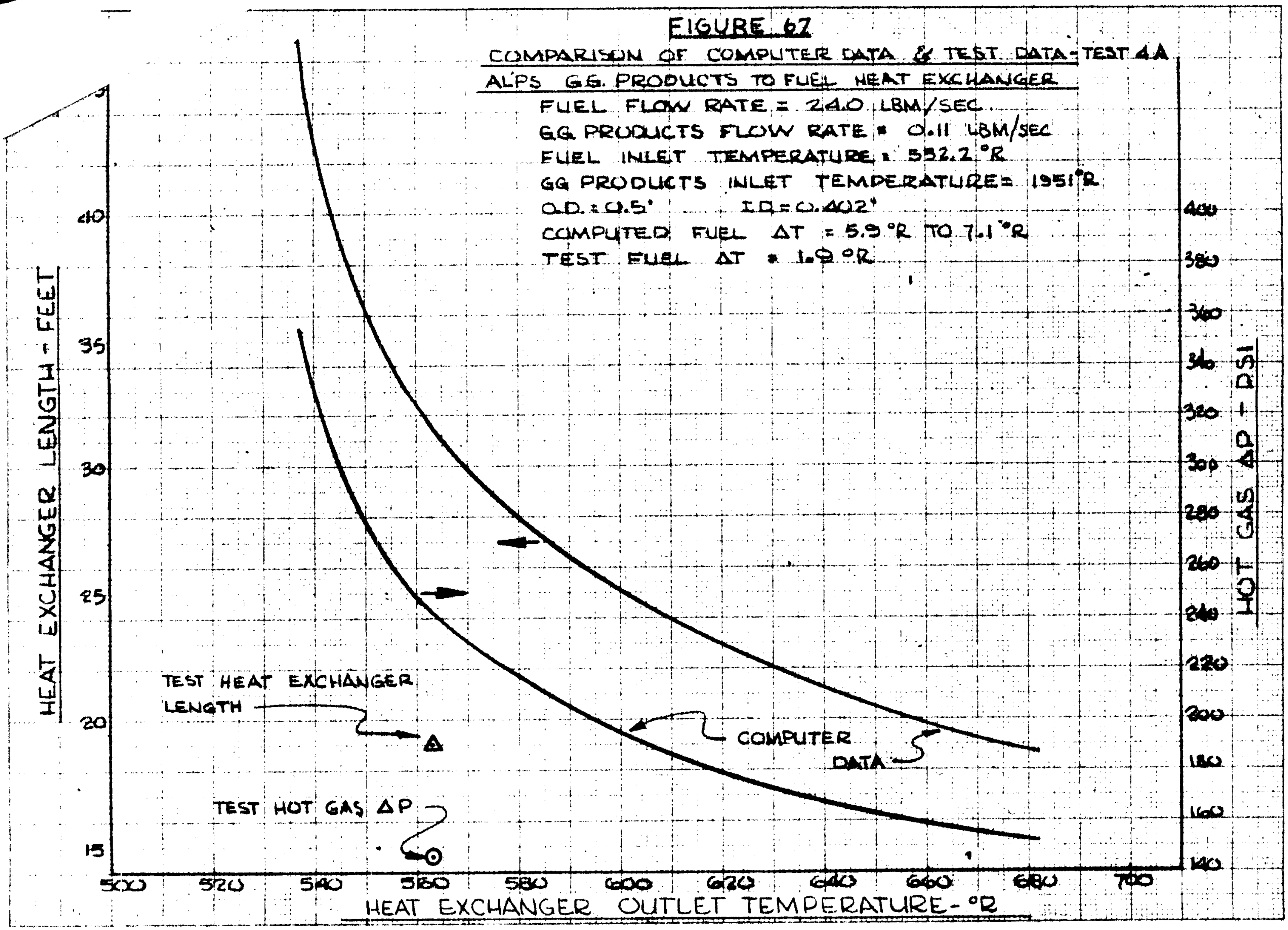


FIGURE 67

COMPARISON OF COMPUTER DATA & TEST DATA - TEST 4A

ALPS G.G. PRODUCTS TO FUEL HEAT EXCHANGER

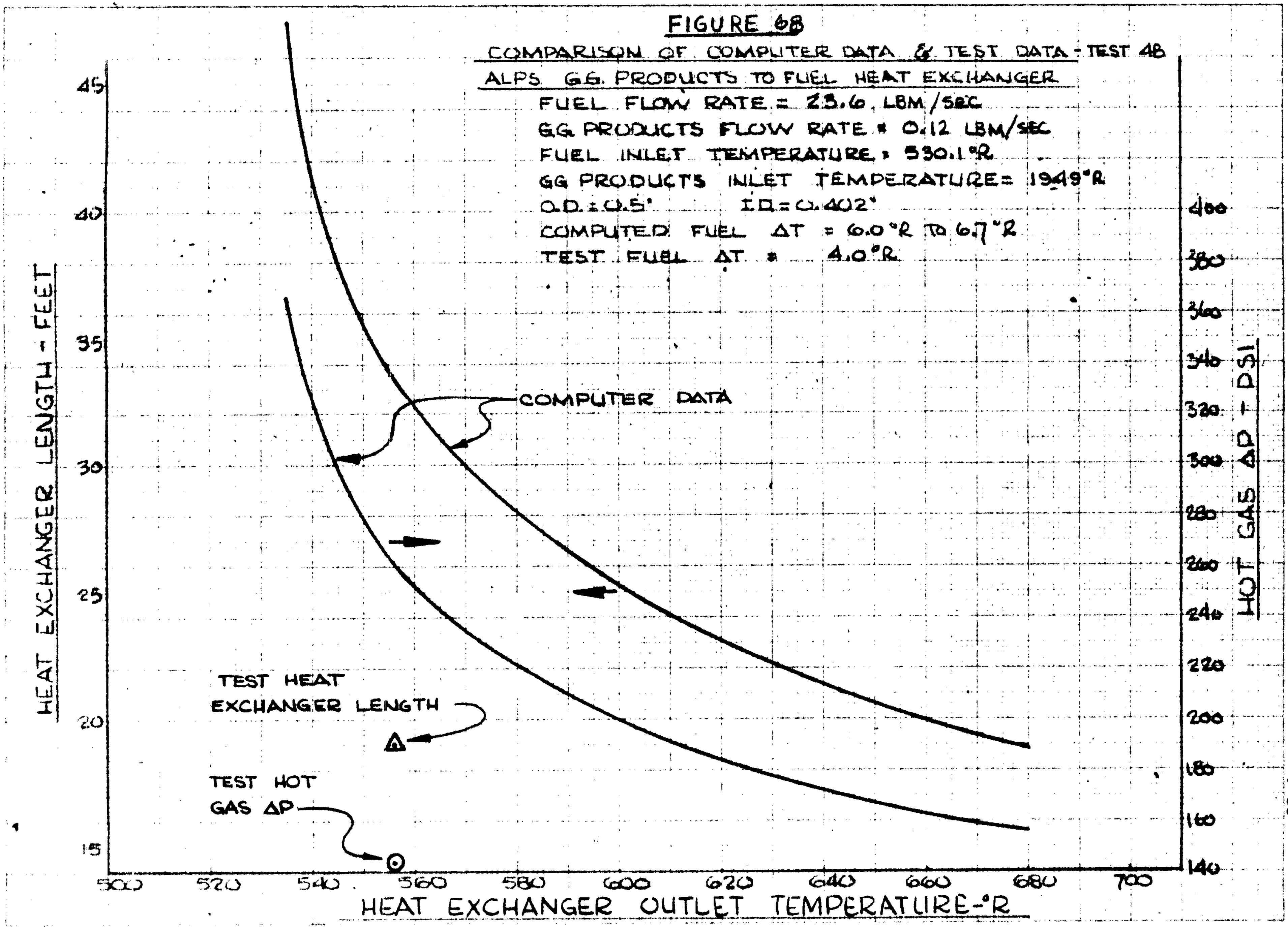


FIGURE 68

COMPARISON OF COMPUTER DATA & TEST DATA - TEST 48

ALPS G.G. PRODUCTS TO FUEL HEAT EXCHANGER

FUEL FLOW RATE = 23.6 LBM/SEC
G.G. PRODUCTS FLOW RATE = 0.12 LBM/SEC
FUEL INLET TEMPERATURE = 530.1°R
GG PRODUCTS INLET TEMPERATURE = 1949°R
O.D. = 0.5"   I.D. = 0.402"
COMPUTED FUEL ΔT = 6.0°R TO 6.7°R
TEST FUEL ΔT = 4.0°R

sample and analysis of the gas generator products would have given the actual chemical composition from which the physical and thermodynamic properties of the gas generator products could have been obtained. Use of the actual physical and thermodynamic properties in the computer program should have helped bring the predicted values closer to the actual test conditions. However, the greatest improvement will be obtained by revising the heat transfer equations in the analytical model to obtain predictions only slightly conservative in nature.

Test 3 was run at a gas generator chamber pressure, Pcgg, of 242 psig and heat exchanger outlet pressure, Pgo, of 152 psig during steady state operation. The associated hot gas flow rate was 0.094 lbm/sec. During Test 4, Pcgg was 280 psig, Pgo was 82 psig, and the flow rate was 0.134 lbm/sec. Pulsing operation was conducted during Tests number 3 and 4, after steady state operation had been achieved. This pulsing was made at 0.25 to 1 cps under the conditions listed above. Typical starting response times while pulsing, as measured from opening of the propellant valve, were as follows:

1) 15 milliseconds to start chamber pressure increase for both tests;
2) 60 milliseconds to 63% of maximum chamber pressure for both tests;
3) 350 milliseconds to 90% of maximum chamber pressure on Test 3;
4) 220 milliseconds to 90% of maximum chamber pressure on Test 4.

Maximum chamber pressure rise rate was approximately 2500 psi/sec during the first 60 milliseconds of operation. After this the rise rate was about 50 psi/sec. Two chamber pressure oscillations were evident during most of the start transients. They were 15-50 psig in magnitude, occurred during the first 80 milliseconds, and were apparently the result of priming the gas generator and ignition transients. The magnitude of these oscillations was a function of shutdown time between pulses. As shut-off time decreased to 800 milliseconds, one oscillation disappeared and the other decreased to 15 psig.

Decay of chamber pressure after closing the propellant valve commenced after about 10 milliseconds and achieved a maximum decay rate of 2500 psi/sec. Fifty percent of maximum chamber pressure was reached in approximately 60 milliseconds after propellant valve closing. These figures are typical for both tests. Ninety percent of maximum P_{GO} was reached in 550 milliseconds on Test 3 and 330 milliseconds on Test 4. Maximum P_{GO} rise and decay rates were about 350 psi/sec on both tests.

Gas generator chamber temperature, measured at the gas generator outlet, stabilized at 1370°F during pulsing compared to 1443°F during steady flow for Test 3. During Test 4, gas generator chamber temperature stabilized in a cycle between 1240°F and 1296°F during pulsing compared to 1462°F during steady flow.

On the basis of these tests, pulsing operation of a monopropellant gas generator is quite feasible. The unit tested, while not designed for pulsing operation, operated satisfactorily at up to 1 cps with two hot gas flow rates, 0.094 lb_m/sec and

0.134 lb_m/sec. A nitrogen purge was used to clear fuel from the gas generator each time it was shut down; this was done to prevent detonation of trapped hydrazine by heat soak-back through the injector. This purge is thought to have contributed to the chamber pressure oscillations at the start of each pulse. The gas generator manufacturer states that a unit designed specifically for pulsing operation would not have this problem and further states that this has been proven by the design and testing of a smaller unit.

C. OPTIMUM SYSTEM SELECTION

The one most important facet of the entire Phase I work was the selection of the most promising pressurization system for the prototype build and test program to be executed during Phase II. Of the eight original systems, six were chosen for preliminary study, analysis, and evaluation (Martin CR-64-80, "Survey of System Concepts," November 1964). Subsequently, a numerical comparison and evaluation of the six systems resulted in the selection of the three most promising systems for more detailed design, analysis, and evaluation (Martin CR-65-6, "System Selection Summary for Advanced Lightweight Pressurization System," January 1965). This detailed study effort included experimental as well as analytical investigations and resulted in the selection discussed herein; additional details concerning this selection are presented in (Martin CR-64-82 (Issue 8), "Monthly Progress Report to Contract NAS 9-3521, June 1965).

1. Criteria for Pressurization System Selection and Optimization

The basis for system selection was previously reported in (Martin CR-65-12, "Criteria for Advanced Lightweight Pressurization System Prototype Selection and Optimization," February 1965). In accordance with that report, the following criteria were considered in the prototype system selection:

Mass
Envelope
Reliability
Minimization of System Start-up Time
Minimization of Pressurization System Leakage
Minimization of Propellant Tank Venting
Cost
Component Availability
Complexity

Ground System Requirements
Storage Time up to Thirty Days

All three systems are discussed in terms of each of the selection criteria in the following section.

2. Comparison of the Three Candidate Systems

a. Mass

One of the basic requirements in developing a new pressurization system for the Apollo Service Propulsion System is a significant reduction in mass, as compared to the present system. Therefore, minimum mass is considered as the most important single criterion for system selection. The pressurization system presently used has a total mass of 1012 lb_m. An optimized, calculated mass of 880 lb_m was derived for purposes of comparison. Masses for the three candidate systems are tabulated in Tables 17, 18, and 19; and are plotted in Figures 69 70, and 71. Tables 17, 18, and 19 show that the mass of all three systems decreases as the thickness of insulation surrounding the helium storage container decreases from 2.0 inches to 0.5 inch. However, the value of insulation thermal conductivity used in these calculations is not considered reliable at thicknesses less than 1.0 inch. In the interest of pursuing a conservative design, only the 2-inch insulation thickness was considered. The storage of helium at either 37°R or 140°R offers considerable mass reductions over the present Apollo system. The savings in mass dissipates rather rapidly at initial storage temperatures above 140°R; therefore, temperatures above

Tank Conditions: Tank Inlet Temperature = 475°R; Propellant Temperature = 500°R

Helium Storage Temp. (°R)	Helium Storage Pressure (psia)	Storage Container Insulation Thickness (inches)	Weight of Helium Loaded, Container, & Insulation (pounds)	Weight of LN_2 or LH_2 Jacket (pounds)	Weight of Valves, Fittings, & Supports (pounds)	Weight of Heat Ex-changers (pounds)	Total System Weight (pounds)	Total System Weight with Jacket (pounds)
300.0	1000	2.0	814.04		71.00	29.0	914.04	
300.0	2000	2.0	584.96		71.00	29.0	684.96	
300.0	3000	2.0	545.06		71.00	29.0	645.06	
300.0	4000	2.0	536.24		71.00	29.0	636.24	
300.0	4000	1.0	526.79		71.00	29.0	626.79	
300.0	4000	0.5	521.67		71.00	29.0	621.67	
140.0	1000	2.0	588.30	102.55	71.00	33.0	692.30	794.85
140.0	2000	2.0	431.87	41.25	71.00	33.0	535.87	577.72
140.0	3000	2.0	410.50	28.03	71.00	33.0	514.50	542.53
140.0	4000	2.0	412.45	22.36	71.00	33.0	516.45	538.81
140.0	4000	1.0	400.46	22.11	71.00	33.0	504.46	526.57
140.0	4000	0.5	388.74	21.69	71.00	33.0	492.74	514.43
60.0	1000	2.0	427.81		71.00	34.0	532.81	
60.0	2000	2.0	346.35		71.00	34.0	451.35	
60.0	2000	1.0	325.89		71.00	34.0	430.89	
60.0	2000	0.5	300.33		71.00	34.0	405.33	
60.0	3000	2.0	353.18		71.00	34.0	458.18	
60.0	4000	2.0	373.35		71.00	34.0	478.35	
37.0	1000	2.0	429.62	45.61	71.00	34.0	534.62	580.23
37.0	2000	2.0	378.92	26.21	71.00	34.0	483.92	510.13
37.0	2000	1.0	348.99	24.80	71.00	34.0	453.99	478.79
37.0	2000	0.5	300.21	21.89	71.00	34.0	405.21	427.10
37.0	3000	2.0	395.93	21.41	71.00	34.0	500.93	522.34
37.0	4000	2.0	422.02	19.03	71.00	34.0	527.02	546.05

Table 17 - Total Weights for System 1

Tank Conditions: Tank Inlet Temperature = 475°R; Propellant Temperature = 500°R
Cascade Container Conditions: P_S = 4000 PSIA, T_S = 530°R, P_f = 450 PSIA

Primary Helium Storage Temp. (°R)	Primary Helium Storage Press.	Primary Container Insulation Thickness	Wt. of Primary Helium Loaded, Container & Insulation (lbs)	Weight of LN_2 or LH_2 Jacket	Weight of Cascade He Loaded and Container (lbs)	Weight of Valves, Fittings, & Supports	Weight of Heat Exchangers (lbs)	Total System Weight	Total Sys. Wt with LN_2 or LH_2 Jacket
300.0	1000	2.0	400.72		338.24	93.2	29.0	861.16	
300.0	2000	2.0	409.24		173.86	93.2	29.0	705.30	
300.0	3000	2.0	423.38		126.48	93.2	29.0	672.06	
300.0	4000	2.0	438.70		101.36	93.2	29.0	662.26	
300.0	4000	1.0	430.81		100.78	93.2	29.0	653.79	
300.0	4000	0.5	427.15		99.62	93.2	29.0	648.97	
140.0	1000	2.0	237.88	42.12	193.02	93.2	33.0	557.10	599.22
140.0	2000	2.0	248.68	24.64	119.95	93.2	33.0	494.83	519.47
140.0	3000	2.0	262.94	18.80	93.54	93.2	33.0	482.68	501.48
140.0	4000	2.0	278.05	15.85	79.77	93.2	33.0	484.02	499.87
140.0	4000	1.0	272.54	15.85	78.69	93.2	33.0	477.43	493.28
140.0	4000	0.5	270.01	15.85	76.71	93.2	33.0	472.92	488.77
60.0	1000	2.0	152.03		104.92	93.2	34.0	384.15	
60.0	2000	2.0	166.30		71.94	93.2	34.0	365.44	
60.0	2000	1.0	161.35		61.12	93.2	34.0	348.67	
60.0	2000	0.5	159.09		55.70	93.2	34.0	341.99	
60.0	3000	2.0	183.44		70.44	93.2	34.0	381.08	
60.0	4000	2.0	200.85		68.10	93.2	34.0	396.15	
37.0	1000	2.0	127.30	15.03	77.82	93.2	34.0	332.32	347.35
37.0	2000	2.0	145.81	11.57	60.07	93.2	34.0	333.08	344.65
37.0	2000	1.0	141.56	11.57	54.12	93.2	34.0	322.88	334.45
37.0	2000	0.5	139.64	11.57	51.01	93.2	34.0	317.85	329.42
37.0	3000	2.0	164.79	10.34	58.91	93.2	34.0	350.90	361.24
37.0	4000	2.0	183.09	9.68	56.70	93.2	34.0	366.99	376.67

Table 18- Total Weights for System 5

Oxidizer Tank Inlet Temperature = 515°R
Fuel Tank Inlet Temperature = 540°R

Oxidizer Temperature = 500°R
Fuel Temperature = 500°R

Helium Storage Temp. (°R)	Helium Storage Press. (psia)	Storage Container Insulation Thickness (inches)	Helium Loaded Storage Container & Insulation Wt.	Weight of LN_2 or LH_2 Jacket (pounds)	N_2H_4 Loaded & Container Weight (pounds)	Gas Generator Weight (pounds)	Weight of Heat Exchangers (pounds)	Valves, Plumbing & Supports Weight (pounds)	Total System Weight (pounds)	Total System Weight with LN_2 or LH_2 Jacket (pounds)
530	1000	0.0	574.50		104.64	8.0	25.5	97.96	810.60	
530	2000	0.0	432.20		104.64	8.0	25.5	97.96	668.30	
530	3000	0.0	409.02		104.64	8.0	25.5	97.96	645.12	
530	4000	0.0	404.45		104.64	8.0	25.5	97.96	640.55	
300	1000	2.0	453.49		104.64	8.0	30.0	97.96	694.09	
300	2000	2.0	324.84		104.64	8.0	30.0	97.96	565.44	
300	3000	2.0	302.18		104.64	8.0	30.0	97.96	542.78	
300	4000	2.0	297.11		104.64	8.0	30.0	97.96	537.71	
300	4000	1.0	290.57		104.64	8.0	30.0	97.96	531.17	
300	4000	0.5	287.0		104.64	8.0	30.0	97.96	527.60	
140	1000	2.0	323.11	56.64	104.64	8.0	32.0	97.96	565.71	622.35
140	2000	2.0	237.33	23.61	104.64	8.0	32.0	97.96	479.93	503.54
140	3000	2.0	225.50	16.45	104.64	8.0	32.0	97.96	468.10	484.55
140	4000	2.0	226.23	13.35	104.64	8.0	32.0	97.96	468.83	482.18
140	4000	1.0	218.46	13.22	104.64	8.0	32.0	97.96	461.06	474.28
140	4000	0.5	210.86	12.94	104.64	8.0	32.0	97.96	453.46	466.40
60	1000	2.0	227.45		104.64	8.0	33.0	97.96	471.05	
60	2000	2.0	185.95		104.64	8.0	33.0	97.96	429.55	
60	2000	1.0	171.92		104.64	8.0	33.0	97.96	415.52	
60	2000	0.5	156.85		104.64	8.0	33.0	97.96	400.45	
60	3000	2.0	188.58		104.64	8.0	33.0	97.96	432.18	
60	4000	2.0	199.18		104.64	8.0	33.0	97.96	442.78	
37	1000	2.0	220.10	24.34	104.64	8.0	33.0	97.96	463.70	488.04
37	2000	2.0	196.19	14.70	104.64	8.0	33.0	97.96	439.79	454.49
37	2000	1.0	178.21	13.89	104.64	8.0	33.0	97.96	421.81	435.70
37	2000	0.5	155.90	12.59	104.64	8.0	33.0	97.96	399.50	412.09
37	3000	2.0	205.99	12.29	104.64	8.0	33.0	97.96	449.59	461.88
37	4000	2.0	220.01	11.04	104.64	8.0	33.0	97.96	463.61	474.65

Table 19 - Total Weights for System 8

FIGURE 69

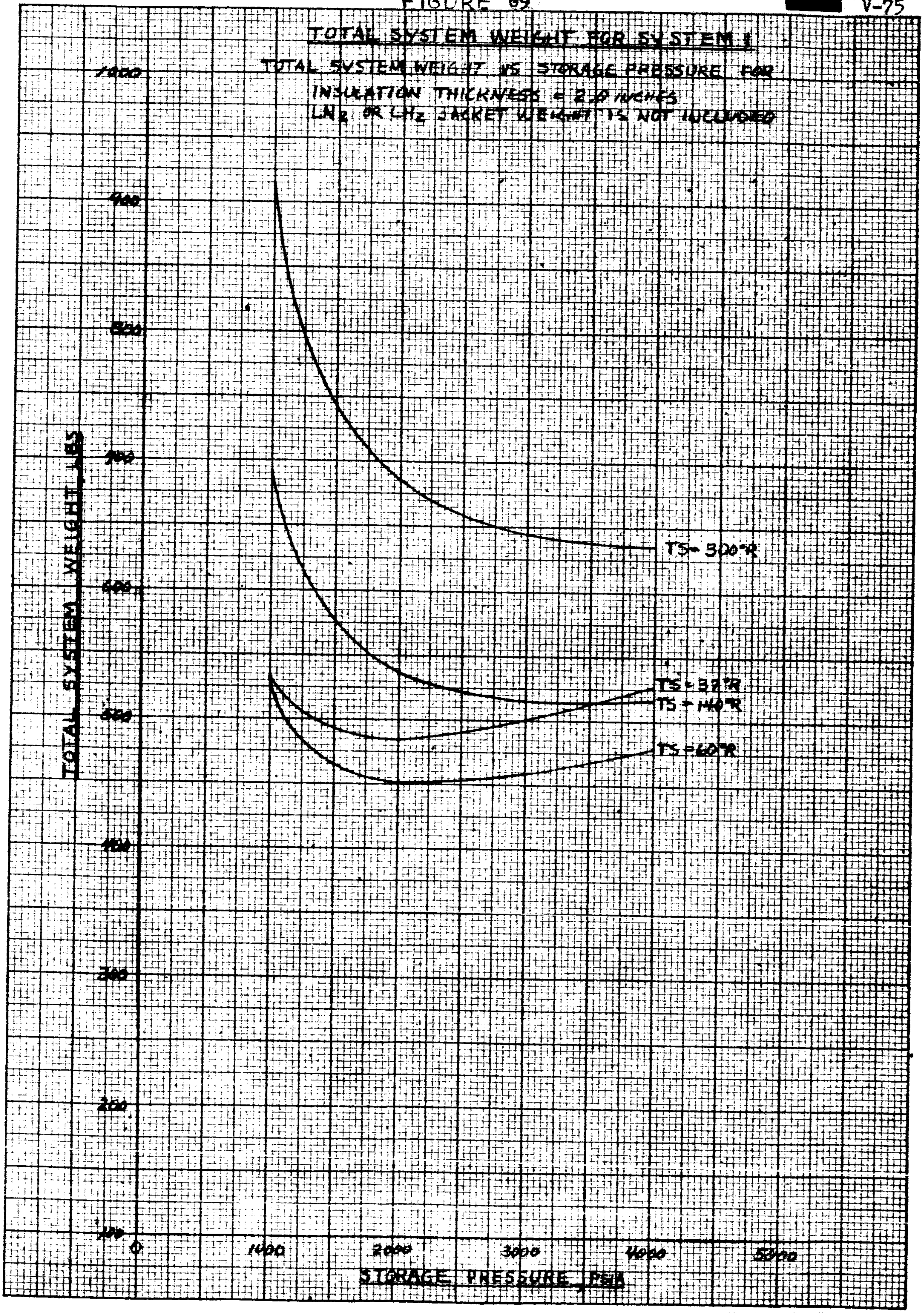
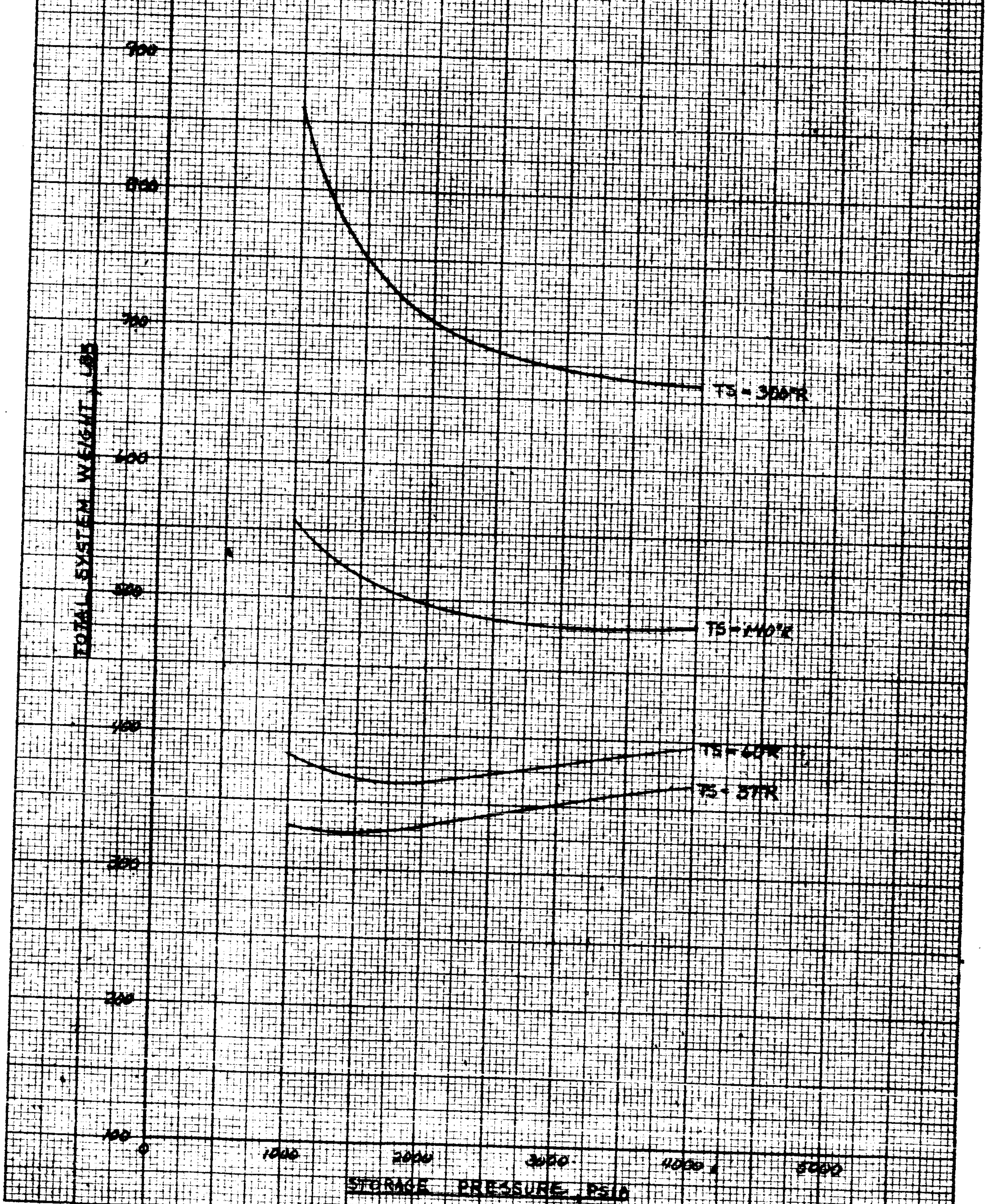

TOTAL SYSTEM WEIGHT FOR SYSTEM I
TOTAL SYSTEM WEIGHT VS STORAGE PRESSURE FOR
INSULATION THICKNESS = 2.0 INCHES
LN_2 OR LH_2 JACKET WEIGHT IS NOT INCLUDED
1000
900
800
700
600
500
400
300
200
100
0
1000
2000
3000
4000
5000
TOTAL SYSTEM WEIGHT, LBS
STORAGE PRESSURE, PSIA
TS = 300°R
TS = 37°R
TS = 140°R
TS = 60°R

FIGURE 20

TOTAL SYSTEM WEIGHTS FOR SYSTEM 5

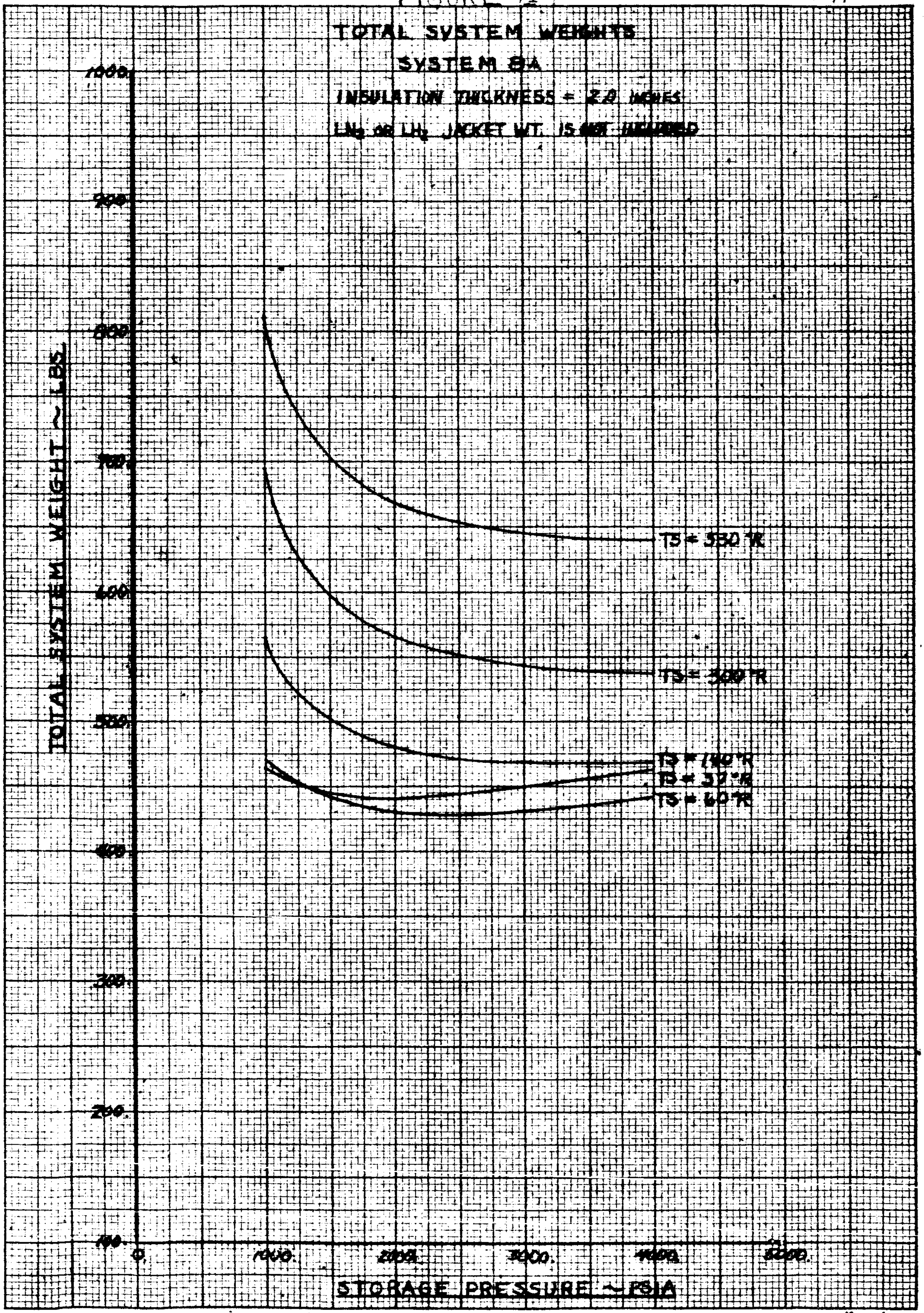
TOTAL SYSTEM WEIGHT VS STORAGE PRESSURE FOR INSULATION THICKNESS = 2.0 INCHES

LN_2 OR LH_2 JACKET WEIGHT IS NOT INCLUDED



K•E 10 X 10 TO THE ½ INCH 359-11 KEUFFEL & ESSER CO. MADE IN U.S.A.

FIGURE 71



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K+E 10 X 10 TO THE ½ INCH 359-11 KEUFFEL & ESSER CO. MADE IN U.S.A.

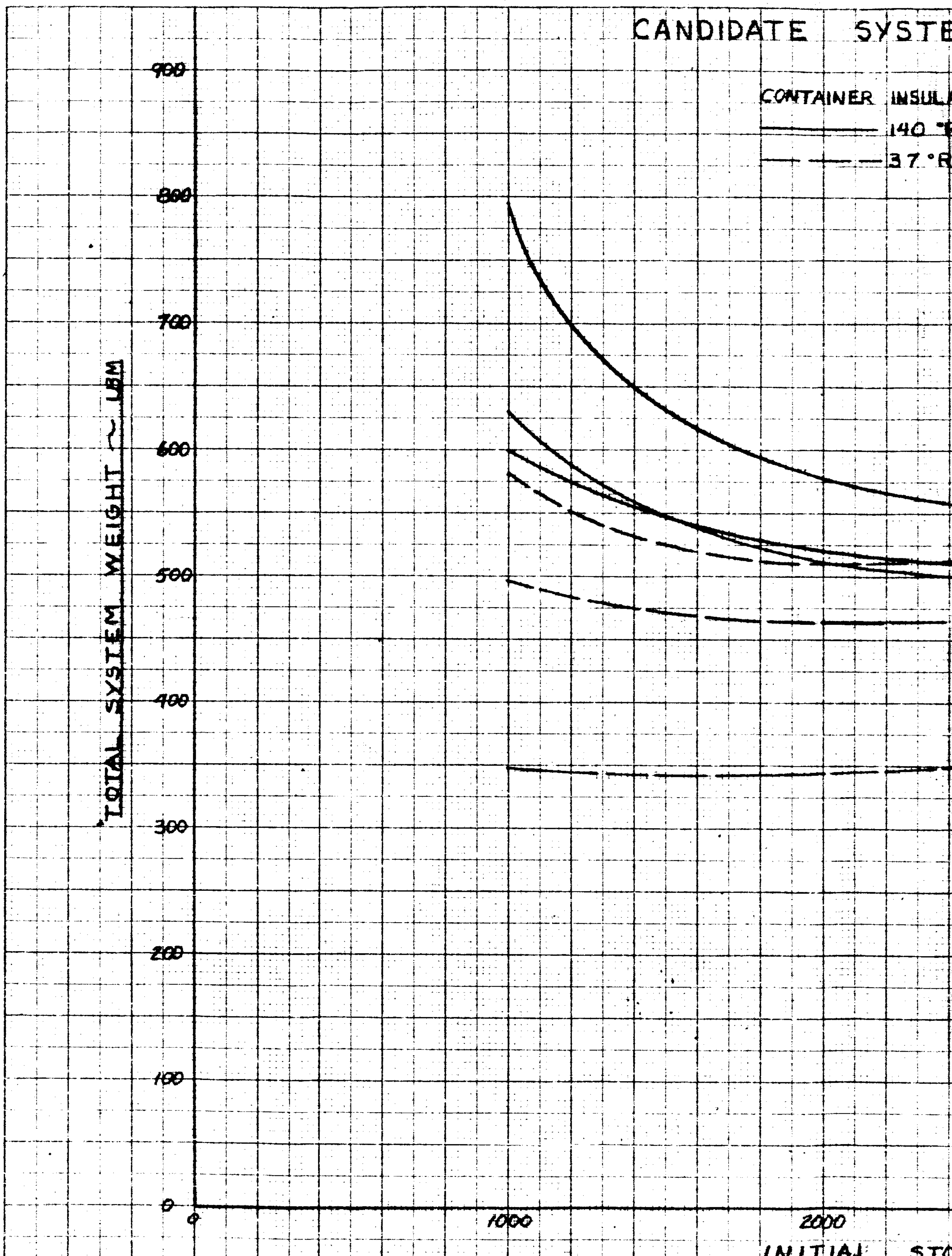
140°R were excluded from further consideration. Initial storage temperatures between 37°R and 140°R are excluded, as discussed under the criterion "Ground System Requirements". Total masses for all three candidate systems at 37°R and 140°R have been plotted as a function of initial helium storage pressure in Figure 72. It is observed that System 5 at 37°R is clearly the optimum system from the minimum mass aspect. Considering the present system optimized mass of 880 lb_m, System 5 offers a potential reduction of 535 lb_m. Table 20 illustrates the potential reduction in mass for each of the candidate systems, at 37°R and 140°R initial storage temperatures.

b. Envelope

All three candidate systems were designed to conform to the geometry limitations of the existing Apollo Service Propulsion System. Furthermore, there is little discernable difference in the overall envelope of all three candidate systems. Therefore, this criterion had no influence upon final system selection.

c. Reliability

Final reliability analyses were completed for the three candidate systems and for the present Apollo SPS pressurization system. Results of these analyses are tabulated in Tables 21, 22, 23 and 24. The effects of all components were considered in this effort. This included vent-relief valves, fill-line disconnect, filters, lines and fittings which were omitted from earlier analyses. Therefore, the reliability numbers given reflect realistic evaluations of the systems. It is noted that



CANDIDATE SYSTE
CONTAINER INSUL
140 °
37 °R
TOTAL SYSTEM WEIGHT ~ LBM
900
800
700
600
500
400
300
200
100
0
0
1000
2000
INITIAL STO

VI-79-1

FIGURE 72

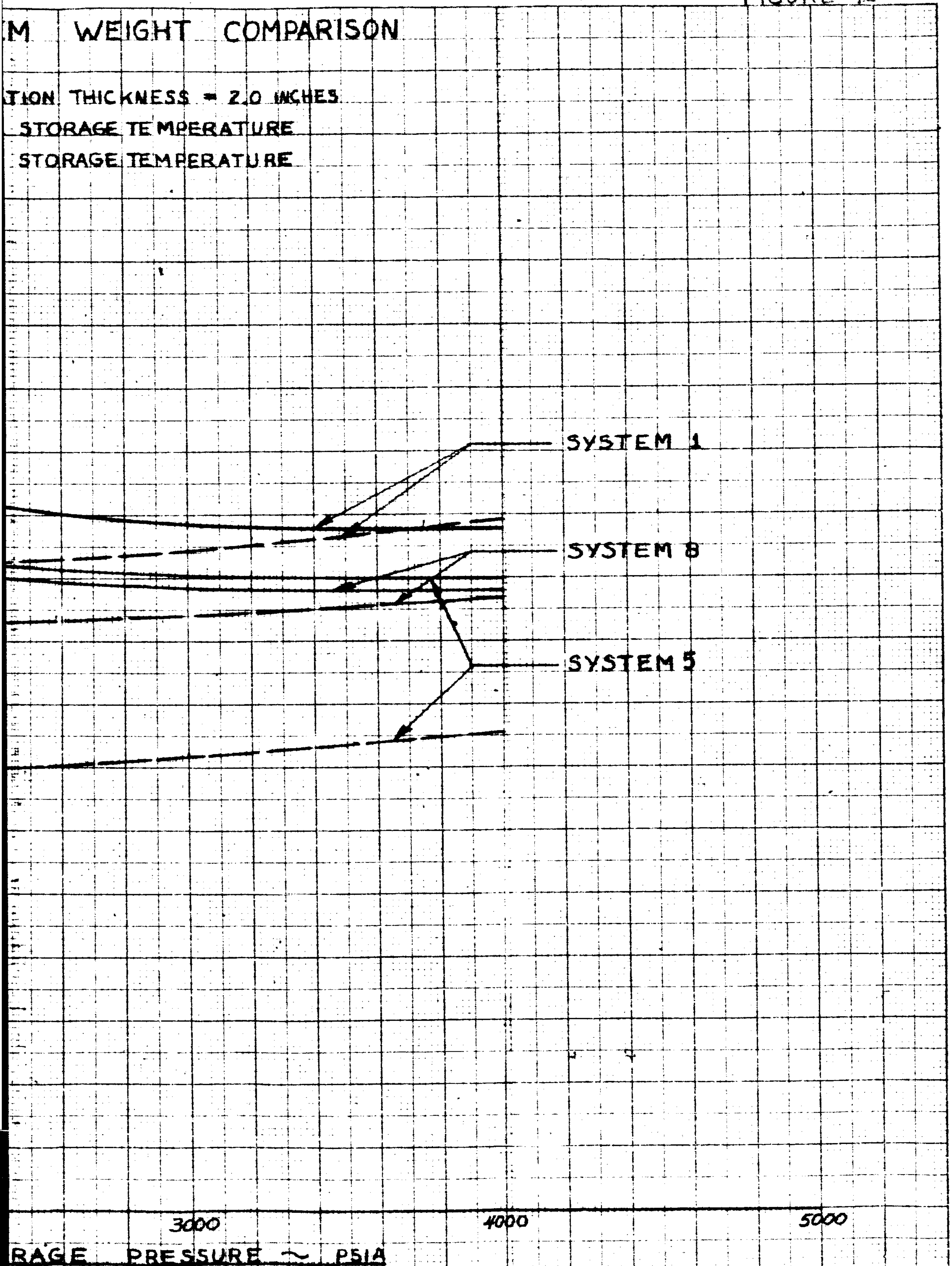


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Table 20
Potential Weight Reduction of the ALPS candidate Systems

Candidate System	Helium Storage Temperature (°R)	Helium Storage Pressure (psia)	Potential Weight Reduction (lbm)
1	140.0	4000.0	341.0
5	140.0	4000.0	380.0
8	140.0	4000.0	398.0
1	37.0	2000.0	369.0
5	37.0	2000.0	535.0
8	37.0	2000.0	426.0

Table 21
Reliability Analysis of System 1

Component	No Of Compon-ents	Failure Mode	Generic Failure Rate Per Item x 10^{-6} Hours	Generic Failure Rate Modifier*	LN of Reliability x 10^{-6} Hrs.
Helium Tank	1	Leak	.07	30.485	2.1339
Pressure Control Valve (Quad)(Fuel)	1 Quad	Fail Closed			.04
Pressure Control Valve (Quad)(Fuel)	1 Quad	Fail Open			.05
Pressure Control Valve (Quad)(Oxid)	1 Quad	Fail Closed			.04
Pressure Control Valve (Quad)(Oxid)	1 Quad	Fail Open			.05
Pressure Switch (Quad)(Fuel)	1 Quad	Fail to Sense Low			.02
Pressure Switch (Quad) (Fuel)	1 Quad	Fail to Sense High			.008
Pressure Switch (Quad) (Oxid)	1 Quad	Fail to Sense Low			.02
Pressure Switch (Quad) (Oxid)	1 Quad	Fail to Sense High			.008
Filter (Helium)	1	Leak	.043	30.485	1.310
Disconnect (Helium Fill) (Manual)	1	Leak	.0005	30.485	.015
Disconnect (Oxidizer Vent) (Manual)	1	Leak	.0005	30.485	.015
Disconnect (Fuel Vent) (Manual)	1	Leak	.0005	30.485	.015
Relief Valve & Diaphragm Assembly	2	Leak			.04
Heat Exchangers	2	All	.6	30.485	36.582
Orifice	2	Plugged	.0001	30.485	.006
Lines & Fittings	21	All	.2	30.485	128.037
			TOTAL		138.390
			Reliability = .999862		

	K_{op}		K_A		
*Boost Phase	250 x .01	x	.1	=	.25
SPS Operate	145 x1.0	x	.1639	=	23.7655
SPS Coast	3 x .01	x	215.6561	=	6.4695

Total . . . . . 30.4850 Generic Failure Rate Modifier

Table 22
Reliability Analysis of System 5

Component	No of Compon-ents	Failure Mode	Generic Failure Rate Per Item x 10^{-6} Hours	Generic Failure Rate Modifier*	LN of Reliability x 10^{-6} Hours
Helium Supply Tank	1	Leak	.07	30.485	2.1339
Helium Cascade Tank	1	Leak	.07	30.485	2.1339
Pressure Control Valve (Quad)(Fuel)	1 Quad	Fail Closed			.04
Pressure Control Valve (Quad)(Fuel)	1 Quad	Fail Open			.05
Pressure Control Valve (Quad)(Oxid)	1 Quad	Fail Closed			.04
Pressure Control Valve (Quad)(Oxid)	1 Quad	Fail Open			.05
Pressure Switch (Quad)(Fuel)	1 Quad	Fail to Sense Low			.02
Pressure Switch (Quad) (Fuel)	1 Quad	Fail to Sense High			.008
Pressure Switch (Quad) (Oxid)	1 Quad	Fail to Sense Low			.02
Pressure Switch (Quad) (Oxid)	1 Quad	Fail to Sense High			.008
Filter (Helium)	1	Leak	.043	30.485	1.310
Disconnect (Hel. Fill) (Manual)	1	Leak	.0005	30.485	.015
Pressure Switch (Quad) (Helium)	1 Quad	Fail to Sense Low			.02
Pressure Switch (Quad) (Helium)	1 Quad	Fail to Sense High			.008
Pressure Control Valves (Quad)(Helium)	1 Quad	Fail Closed			.04
Pressure Control Valves (Quad)(Helium)	1 Quad	Fail Open			.05
Filter (Helium, Cascade)	1	Leak	.043	30.485	1.310
Disconnect (Helium Fill Cascade) (Manual)	1	Leak	.0005	30.485	.015
Relief Valve & Diaphragm Assembly	2	Leak			.04
Heat Exchangers	2	All	.6	30.485	36.582
Orifice	2	Plugged	.0001	30.485	.006
Disconnect (Oxid. Vent)((Manual)	1	Leak	.0005		.015
Disconnect (Fuel Vent) (Manual)	1	Leak	.0005		.015
Lines & Fittings	29	All	.2	30.485	176.813
			TOTAL		220.742
			Reliability = 999780		

	K_{op}		K_A		t		
*Boost Phase	250	x	.01	x	.1	=	.25
SPS Operate	145	x	1.0	x	.1639	=	23.7655
SPS Coast	3	x	.01	x	215.6561	=	6.4695

Total = 30.4850 Generic Failure Rate Modification

Table 23
Reliability Analysis of System 8

Component	No. of Components	Failure Mode	Generic Failure Rate Per Item x 10^{-6} Hours	Generic Failure Rate Modifier*	LN of Reliability x 10^{-6} Hours
Helium Tank	1	Leak	.07	30.485	2.1337
GG Propellant Tank	1	Leak	.07	30.485	2.1337
Pressure Control Valve (Quad) (Oxid)	1 Quad	Fail Closed			.05
Pressure Control Valve (Quad) (Oxid)	1 Quad	Fail Open			.04
Pressure Switch (Quad) (Oxid)	1 Quad	Fail to Sense Low			.02
Pressure Switch (Quad) (Oxid)	1 Quad	Fail to Sense High			.008
Pressure Switch (Quad) (Fuel)	1 Quad	Fail to Sense Low			.02
Pressure Switch (Quad) (Fuel)	1 Quad	Fail to Sense High			.008
Disconnect (Helium Fill) (Manual)	1	Leak	.0005	30.485	.015
Pressure Control Valve (Quad) (GG Propellant Tk)	1	Fail Closed			.05
Pressure Control Valve (Quad) (GG Prop. Tank)	1	Fail Open			.04
Filters (Helium) (GG Fuel)	2	Leak	.0001	30.485	.006
Disconnect (Propellant Fill and Drain)	1	Leak	.0005	30.485	.015
Propellant Control Valves GG (Quad)	1 Quad	Fail Closed			.05
Propellant Control Valves GG (Quad)	1 Quad	Fail Open			.04
Gas Generator (Fuel)	1	All	2.2	30.485	67.0670
Disconnect (Oxid. Vent) (Manual)	1	Leak	.0005	30.485	.015
Disconnect (Fuel Vent) (Manual)	1	Leak	.0005	30.485	.015
Relief Valve & Diaphragm Assembly	2	Leak			.04

(continued on next page)

Table 23
Reliability Analysis of System 8 (continued)

Component	No. of Compon-ents	Failure Mode	Generic Failure Rate per Item x 10^{-6} Hours	Generic Failure Rate Modifier*	LN of Reliability x 10^{-6} Hours
Bellows (GG Propellant)	1	Leak	2.24	30.485	68.286
Pressure Switch (Quad) (GG Propellant)	1 Quad	Fail to Sense Low			.02
Pressure Switch (Quad) (GG Propellant)	1 Quad	Fail to Sense High			.008
Lines and Fittings	24	All	.2	30.485	146.328
			TOTAL		286.408
			Reliability = .999714		

	K_{op}		K_A		t		
*Boost Phase	250	x	.01	x	.1	=	.25
SPS Operate	145	x	.7	x	.16	=	16.20
SPS Coast	3	x	.01	x	215.7	=	6.47
			Total				30.485

Table 24
Reliability Analysis of Present Apollo SPS Pressurization System

Component	No. of Compon-ents	Failure Mode	Generic Failure Rate per Item x 10^{-6} Hours	Generic Failure Rate Modifier*	LN of Reliability x 10^{-6} Hours
Helium Tanks	2	Leak	.07	30.485	4.2678
Helium Valve and Regulator Assembly	Redund	Fail Open			.0000004
Helium Valve and Regulator Assembly	Redund	Fail Closed			.02
Check Valve Assembly	2 Quads	Fail Open			.08
Check Valve Assembly	2 Quads	Fail Closed			.001
Heat Exchangers	2	All	.6	30.485	36.582
Relief Valve and Diaphragm Assembly	2	Fail Open			.04
Helium Supply Disconnect	1	Leak	.0005		.015
Oxidizer Fill and Drain Disconnect	1	Leak	.0005		.015
Fuel Fill and Drain Disconnect	1	Leak	.0005		.015
Oxidizer Vent Disconnect	1	Leak	.0005		.015
Fuel Vent Disconnect	1	Leak	.0005		.015
Filter Helium	1	Leak	.043	30.485	1.310
Lines and Fittings	37	Leak	.2	30.485	225.589
			TOTAL		267.964
			Reliability = .999733		

	K_{op}		K_A		t		
*Boost Phase	250	x	.01	x	.1	=	.25
SPS Operate	145	x	.7	x	.16	=	16.20
SPS Coast	3	x	.01	x	215.7	=	6.47
			Total				30.485

the greatest probable sources of failure in all systems are the lines and fittings. This is due to the fact that the unreliable individual components such as valves, regulators, and pressure switches, have been assembled into redundant units which diminishes considerably the possibility of total functional failures. To eliminate the possibility that the effect of lines and fittings could obscure other important comparative features of candidate system reliability, an analysis was made which omitted lines and fittings from consideration. These results, presented in Table 25 show that System 1 is slightly higher in reliability than the present Apollo system and System 8 is slightly lower.

Table 25
Results of System Reliability Studies

System	Total Reliability	Reliability Without Lines and Fittings
1	.999862	.999990
5	.999780	.999946
8	.999714	.999860
Apollo	.999733	.999948

d. <u>Minimization of System Startup Time</u>

The pressurization system selected should add no appreciable complexity to the Apollo SPS start sequence nor impose time lags which necessitate anticipation of start operations. All three candidate systems were equally advantageous in this respect. None required more than a single command to initiate operation and all were instantly responsive (within the 20 - 50 millisecond time required to actuate a normal solenoid valve). This criterion was therefore not influential in selection of the optimum system.

e. Minimization of Pressurization System Leakage

Since all candidate systems contain the same types of components, the comparison of leakage characteristics of the systems was resolved by a count of all pressurized gas lines which have the potential of leaking gas from the system. The system with the greatest number of lines was considered the least desirable. On this basis, System 1, with 21 lines, was the minimum leak system, System 8, with 24 lines, was second best, and System 5, with 29 lines, ranked third. Since most of the lines will be welded together in the flight system, it is not considered that the criterion of leakage has a strong influence on system selection.

f. Minimization of Propellant Tank Venting

Extensive propellant tank thermodynamic analyses were performed using the design mission profiles and heating data as supplied by NASA-MSC. These studies indicated that the maximum operating pressures of the propellant tank would not be exceeded during the mission, regardless of which of the three candidate pressurization systems were used. Since propellant tank venting is not indicated all three candidate systems are considered to be equally acceptable.

g. Cost

The estimated hardware cost figures for each of the three candidate systems are presented in Table 26. The tabulations include both development and purchase costs for the components.

Table 26
Estimated Flight Qualified Component Costs
(For System Comparison Only)

System 1

Component	No. of Each	Dev. & Qualification Costs	Hardware Costs Per Vehicle
Solenoid Valve	8	$117,500	$22,400
Pressure Switch	2 Quads	100,000	7,200
Disconnect	5	-	2,500
Heat Exchanger	2	100,000	8,000
Filter	1	45,400	318
Jacketed Helium Tank	1	250,000	30,000
Relief Valve	3	200,000	22,500
Lines & Fittings	-	-	5,000
TOTAL		$812,900	$97,918

System 5

Component	No. of Each	Dev. & Qualification Costs	Hardware Costs Per Vehicle
Solenoid Valve	12	$117,500	$33,600
Pressure Switch	3 Quads	100,000	10,800
Disconnect	6	-	3,000
Heat Exchanger	2	100,000	8,000
Filter	1	45,400	318
Jacketed Helium Tank	1	250,000	30,000
Helium Tank	1	132,000	12,000
Relief Valve	3	200,000	22,500
Lines & Fittings	-	-	5,000
TOTAL		$944,900	$125,218

Table 26 (continued)

System 8

Component	No. of Each	Dev. & Qualification Costs	Hardware Costs Per Vehicle
Solenoid Valve	12	$117,500	$33,600
Pressure Switch	3 Quads	100,000	10,800
Disconnect	6	-	3,000
Heat Exchanger	2	100,000	8,000
Filter	2	45,400	636
Jacketed Helium Tank	1	250,000	30,000
Positive Expulsion Tank	1	200,000	10,000
Gas Generator	1	400,000	20,000
Relief Valve	4	200,000	30,000
Lines & Fittings	-	-	5,000
TOTAL		$1,412,900	$151,036

These estimates are based upon actual costs incurred during the Titan III Transtage procurement program, modified to reflect the variations in component requirements and applications. Therefore, the costs are believed to be representative of man-rated, flight-qualified hardware. However, this information is presented for purposes of comparison only and should not be construed as a firm quote.

System 1 is definitely the least expensive of the three and System 8 is the most expensive, being about 50% higher in cost. System 5 is about in the middle, 15% above System 1 in development and qualification costs, and 30% higher in purchase costs per vehicle.

h. Availability

All systems can be developed in approximately 21 months. There are no development span differences between the systems.

i. Complexity

Complexity, as related to system selection, has two connotations:

1) The extent to which a candidate system affects the design and operation of other existing Apollo SPS subsystems, and
2) the inherent complexity of the pressurization system itself, as determined by the total number of components.

Regarding item 1) above, the only existing subsystem affected by any of the candidate systems is the electrical power supply. It is estimated that the solenoid valves used will require power at rates of 0.5 KW for System 1 and .75 KW for Systems 5 and 8. Electrical power is used only during periods of main engine operation, so the total power requirements are about .083 KW-hour for

System 1, and .125 KW-hour for Systems 5 and 8. Voltage required is 28VDC for all three systems.

The total number of working components is 27 for System 1, 38 for System 5, and 39 for System 8. Evaluation of both aspects of complexity, therefore, indicate that System 1 is the most desirable with Systems 5 and 8 being about equal in ranking.

j. Ground System Requirements

Ground system requirements are established by a helium loading time and a ten-hour hold capability. Also, changes to the present ground system requirements should be minimized. The three candidate systems were compared on the basis of these three requirements. The following is a summary of the comparison of the three candidate systems.

1) Helium Loading Requirement

Helium loading time was estimated for the three candidate systems at two storage conditions (Table 27). The helium loading time was defined as the sum of the actual loading time and the time required to cool the helium and the sphere to the storage conditions. For a 140°R storage temperature and 4000 psia storage pressure, the loading times were about the same. System 8 had the least time required for loading and System 1 the greatest. The difference between System 8 and System 1 loading time, however, was only three minutes. For a 37°R storage temperature and a 2000 psia storage pressure, the helium loading time for System 5 was greater than the loading times for Systems 1 and 8. The loading time for System 8 was only three minutes less than that for System 1. On the basis of this comparison, System 8 has the best loading times

Table 27
Results of Helium Loading and Ten-Hour Hold Analysis
Insulation Thickness = 2.0 Inches

Candidate System	Helium Storage Temperature (°R)	Helium Storage Pressure (psia)	Mass of Helium Loaded (lbm)	Helium Loading Rate (lbm/sec)	Helium Loading Temperature (°R)	Estimated Loading Time (minutes)	Wt. of LN_2 or LH_2 Required For Loading (lbm)	Wt. of LN_2 or LH_2 Required For 10 Hour Hold (lbm)
1	140	4000	106.89	.02969	530	104	(LN_2) 766	(LN_2) 88.4
5	140	4000	71.70	.0199	530	101.5	(LN_2) 513	(LN_2) 68.8
8	140	4000	58.14	.0161	530	101	(LN_2) 417	(LN_2) 60.3
1	37	2000	189.30	.0523	530	79	(LH_2) 665	(LH_2) 55.0
5	37	2000	71.70	.0100	530	133	(LH_2) 258	(LH_2) 30.0
8	37	2000	96.90	.0269	530	76	(LH_2) 349	(LH_2) 36.0

(by a slight amount) for both storage conditions. The amount of coolant (LN_2 or LH_2) required to cool the helium and helium sphere down to the storage conditions during and after the loading process was also evaluated. For a 140°R storage temperature and 4000 psia storage pressure, System 8 required 349 lbm of LN_2 less than System 1 and 96 lbm less than System 5. For a 37°R storage temperature and a 2000 psia storage pressure, System 5 used less LH_2 coolant than either Systems 1 or 8. System 8 has the more desirable helium loading requirements for the 140°R, 4000 psia storage conditions while System 5 is more advantageous at storage conditions of 37°R and 2000 psia.

2) Ten-Hour Hold Capability

The capability of each candidate system to maintain a given storage condition for ten hours was evaluated. The amount of coolant (LN_2 or LH_2) required to maintain the given storage conditions was calculated for each candidate system. For a 140°R storage temperature and a 4000 psia storage pressure, System 8 required the least amount of LN_2. System 5 required about 8 lbm more LN_2 than System 8 and System 1 required about 28 lbm more LN_2 than System 8. For storage at 37°R and 2000 psia, System 5 required the least amount of LH_2. System 8 required 6 lbm more LH_2 than System 5 and System 1 required 25 lbm more LH_2 than System 5. For a ten-hour hold capability, System 8 is more desirable for 140°R storage temperature and 4000 psia storage pressure, and System 5 is more desirable for 37°R storage temperature and 2000 psia storage pressure.

3) Other GSE Requirements

In addition to the helium, LN_2, and LH_2 requirements discussed above, System 8 also requires a 98.4 lbm of hydrazine to service the gas generator propellant tank. This is not a large amount but it does require an additional ground supply system which is not needed for Systems 1 and 5.

It is concluded that Systems 1 and 5 have the same types of ground servicing requirements. System 8 also has the same requirements plus the additional requirement of a hydrazine supply. The time required to load the system does not vary significantly - the minimum being 79 minutes for System 8 at 37°R and 2000 psia and the maximum being 133 minutes for System 5 at the same storage conditions.

k. Storage Time Up to Thirty Days

At the conclusion of the detailed design and analysis of the three candidate systems, each system was subjected to an analytical simulation of a thirty-day mission. Two thirty-day mission profiles were considered. The first considered a mission identical to the existing nine-day design with the addition of a twenty-one day coast at the end of the fourth burn period (lunar orbit insertion). This mission profile is referred to as Mission Plan A. The second, referred to as Mission Plan B, was a thirty-day earth orbital coasting with 50% propellant mass loaded followed by a single main engine burn to propellant depletion. The three systems were subjected to the analytical simulation of the two thirty-day missions in order to determine which system was most adaptable to additional missions. The results of this analytical simulation for the three candidate systems is discussed below.

1) Thirty-Day Mission Plan A

The analytical simulation for Mission Plan A used the identical pressurant storage system configuration as the system designed for the nine-day mission. The amount of helium loaded and used were also the same as those required for the nine-day mission. The primary question that the analytical simulation answered was whether the pressure in the helium sphere would exceed the limit operating pressure during the extended fourth coast. Table 28 shows the maximum helium pressures obtained during Mission Plan A simulation, the helium temperature at the maximum pressure, the limit operating pressure at maximum conditions minus maximum pressure, and the time at which maximum pressure was obtained. The results show that the maximum pressures obtained for all three candidate systems at all the storage pressures and temperatures were considerably lower than the limit operating pressure for each sphere. These maximum pressures all occurred at the end of the first coast. Figure 73 shows the limit operating pressures as a function of temperature. On the basis of these results, all three systems are equally adaptable to the thirty-day Mission Plan A.

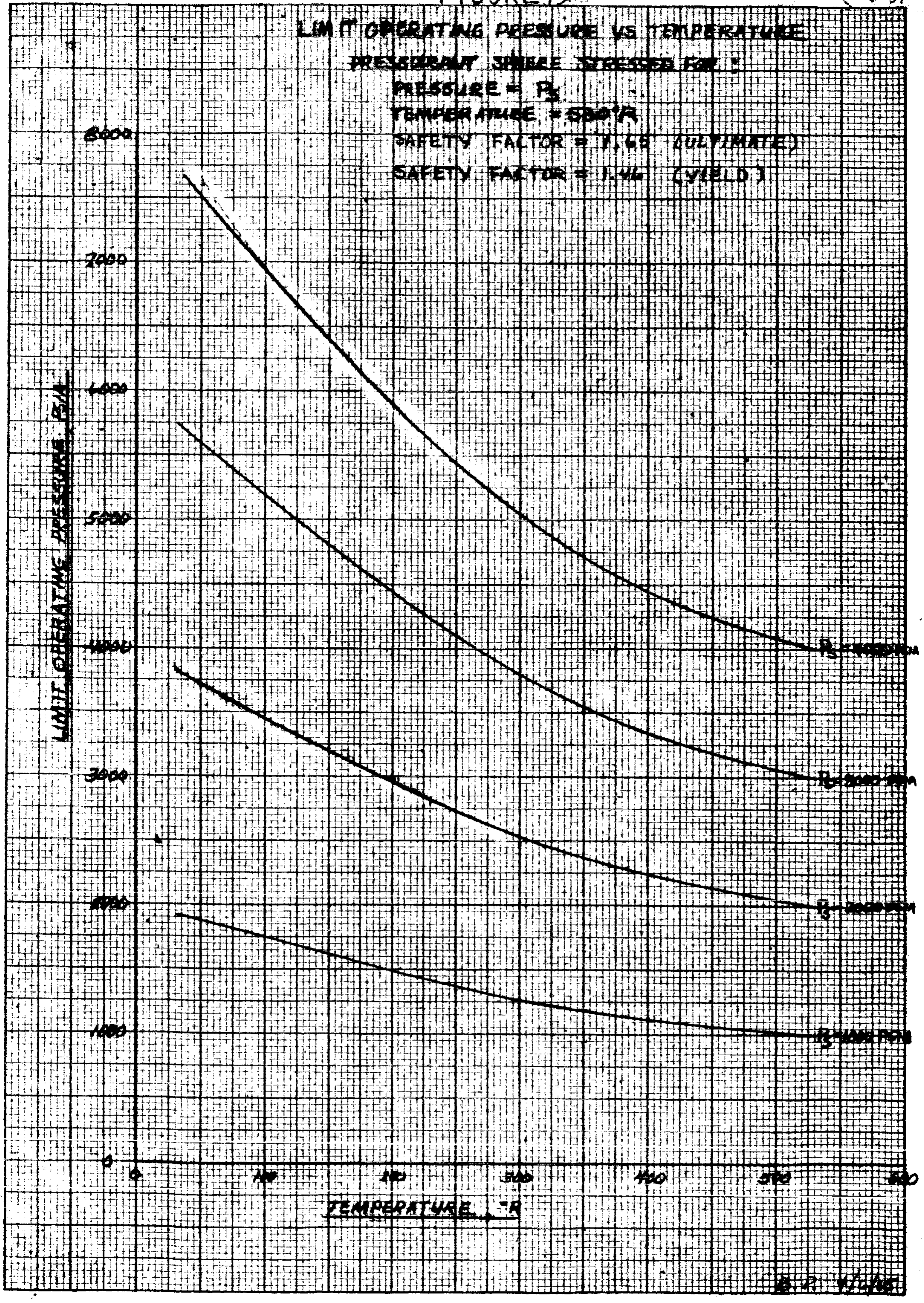
2) Thirty-Day Mission Plan B

The analytical simulation for Mission Plan B also used the identical pressurant storage system configuration as the system designed for the nine-day mission. The helium usage requirement changed since the propellant loaded was 50% of

Table 28
Results of the Analytical Simulation for Storage Time up to Thirty Days

Mission Plan	Candidate System	Helium Storage Temp.	Helium Storage Press.	Heating Rate at Max. Press.	Maximum Helium Press.	Helium Temp. at Max. Press.	Limit Operating Pressure Minus Max. Pressure	Mission Time For Maximum Pressure
A	1	140.0°R	4000 psia	3.0564 $\frac{btu}{hr}$	4022.0 psia	140.73°R	2478.0 psia	End of 1st Coast
	5	140.0°R	4000	2.664	4030.02	141.00	2469.98	End of 1st Coast
	8	140.0	4000	2.5488	4033.75	141.12	2466.25	End of 1st Coast
	1	37.0	2000	3.8376	2034.37	37.61	1735.63	End of 1st Coast
	5	37.0	2000	2.916	2072.07	38.28	1697.93	End of 1st Coast
	8	37.0	2000	3.1572	2055.24	37.98	1714.76	End of 1st Coast
B	1	140.0	2500	2.8224	3072.06	170.29	3127.94	End of 1st Coast
	5	140.0	1500	2.340	2133.57	196.31	3796.43	End of 1st Coast
	8	140.0	2500	2.2752	3345.92	184.84	2704.08	End of 1st Coast
	1	37.0	1250	3.6612	2182.57	60.33	1477.43	End of 1st Coast
	5	37.0	550	2.520	1968.19	112.64	1411.81	End of 1st Coast
	8	37.0	1320	2.9448	2821.71	73.09	778.29	End of 1st Coast

FIGURE 73


LIMIT OPERATING PRESSURE VS TEMPERATURE
PRESSURANT SPHERE STRESSED FOR :
PRESSURE = P_S
TEMPERATURE = 500°R
SAFETY FACTOR = 1.65 (ULTIMATE)
SAFETY FACTOR = 1.46 (YIELD)
LIMIT OPERATING PRESSURE, PSIA
8000
7000
6000
5000
4000
3000
2000
1000
0
0
100
200
300
400
500
600
TEMPERATURE, °R
P_S = 4000 PSIA
P_S = 3000 PSIA
P_S = 2000 PSIA
P_S = 1000 PSIA
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the propellant loaded for the nine-day mission. The helium usage was smaller than that for the nine-day mission. Due to the decrease in helium usage, the helium loaded and the storage pressures were also smaller than the values used in the nine-day mission since identical sphere volumes were used. Again, the primary question that the analytical simulation answered was whether the pressure in the helium sphere would exceed the limit operating pressure during the thirty-day coast period. Table 28 shows the results of the analytical simulation for Mission Plan B. The maximum helium pressures obtained at the end of the thirty-day coast were considerably below the limit operating pressures for each sphere and, therefore, the three candidate systems are adaptable to the thirty-day Mission Plan B. System 5 is more desirable than Systems 1 and 8 for a helium storage temperature of 140°R since the difference between the limit operating pressure and maximum pressure is greater for System 5 than for either Systems 1 or 8. For a storage temperature of 37°R, System 1 is slightly more desirable than System 5 since the difference between limit operating and maximum pressures for System 1 is approximately 65 psi greater than for System 5.

3. Selection of the Optimum Pressurization System

The pertinent results of the system evaluation effort are summarized in Table 29. Examination of this table reveals that significant differences in the three candidate systems are seen in only three areas; mass, cost, and complexity (as measured by total number of components). The criteria of minimum system mass is definitely in favor of system 5, with systems 8 and 1 following in that order. The mass savings are 535 lb_m for system 5, 426 lb_m for system 8, and 370 lb_m for system 1; as compared to the calculated optimum mass of 880 lb_m for the existing Apollo system. Component development and qualification costs are a minimum for system 1; system 5 is about 17% higher; and system 8 is about 28% higher than system 1. Hardware costs per vehicle are again a minimum for system 1, with system 5 being 20% higher and system 8 being 43% higher. System 1 is the least complex by component count, with systems 5 and 8 being equal. However, the additional components required in system 5 are combined in various series - parallel redundant units; which when related to the important criterion of reliability, imposes only a very small penalty upon the system. It is noted that both system 1 and system 5 are compatible with the present Apollo system reliability, while system 8 is less reliable than the present system. It is concluded that the greater savings in mass afforded by system 5 is more significant than the small variations found in evaluation of the other criteria. The Martin Company, therefore, selects system 5, with helium pressurant stored at 37°R, to fulfill the requirements of an advanced lightweight pressurization system for the current Apollo Service Propulsion System.

Table 29 - System Comparison Summary

Comparison Catagory	Apollo	System 1	System 5	System 8	Remarks
1. Mass	880 lbm	510.13 lbm	344.65 lbm	454.49 lbm	Calculated Optimum weights for each system.
2. Envelope	- NO SIGNIFICANT DIFFERENCE -				All within Apollo limitations.
3. Reliability	0.999733	0.999862	0.999780	0.999714	
4. System Start-up Time	- NO SIGNIFICANT DIFFERENCE -				All systems startup time in milli-seconds.
5. System Leakage	- NO SIGNIFICANT DIFFERENCE -				With major portion of system using welded joints leakage becomes in-significant.
6. Propellant Tank Venting	NONE	NONE	NONE	NONE	No venting required from systems studied.
7. Cost Ratios					Cost ratios of systems studied to existing Apollo system.
Dev. & Qual.	1.0	0.9	1.05	1.15	
Cost/Veh.	1.0	1.40	1.68	2.00	
8. System Avail-ability	Current	21 mo.	21 mo.	21 mo.	Hardware Development Time
9. Complexity Additional Pwr. Reqmt.					
Rate	None	0.5 KW	0.75 KW	0.75 KW	Power Requirements Above Existing Apollo System
Total	None	0.083 KWH	0.125 KWH	0.125 KWH	
No. of working compnts.	22	27	38	39	

Table 29 - System Comparison Summary

Comparison Catagory	Apollo	System 1	System 5	System 8	Remarks
10. Grd. Sys. Reqrmts.					
He. Ld. Time	Unkn.	79 min.	133 min.	76 min.	Calculated - not verified by test
10 hr. Hold	None	55 lbm	30 lbm	36 lbm	Amount of coolant (LH_2) required to maintain storage conditions (calculated).
Other	- NO DIFFERENCE -			Hydrazine Loading	Equipment required to load 98# of hydrazine for G.G. propellant tank.
11. 30 Day Storage Capability					
Plan A	- NO SIGNIFICANT DIFFERENCE				All Systems Adaptable
Plan B	- NO SIGNIFICANT DIFFERENCE				All Systems Adaptable

VI. DESIGN SUMMARY OF SELECTED SYSTEM

There are certain aspects of the design of the selected system (system 5) which warrant additional discussion. These areas include design of the primary storage tank, heat leak considerations, and use of existing Apollo components.

A. PRIMARY STORAGE TANK DESIGN

The design installation drawings of the selected system (IAB-6002276, six sheets) shows the proposed approach to design of the primary tank. A simplified drawing of the sphere is shown in Figure 74. A two-inch thick layer of NRC-2 superinsulation (78 sheets per inch) encloses the outer surface of the vessel, providing protection from environmental thermal radiation. The area immediately surrounding the support bosses is filled with foam insulation (polyurethene or polystyrene) as shown. The support tubing and connecting helium and liquid hydrogen supply lines will be covered with superinsulation to minimize heat leak into the vessel.

The pressure vessel and liquid hydrogen jacket are fabricated from titanium alloy 6AL-4V; the extra low intersticial (ELI) grade with impurities sufficiently suppressed to be suitable for use at liquid hydrogen temperature will be used.

The liquid hydrogen jacket is included to provide the required ten-hour ground hold capability, and for precooling the vessel prior to loading the helium. Liquid hydrogen is circulated through the jacket at all times during the cooldown and loading process to establish the required initial storage conditions (37°R and 2000 psia). The jacket is a non-structural unit, sized to contain sufficient liquid hydrogen to sustain normal heat leak during the ten-hour ground hold. If it is required to extend the ground hold period further

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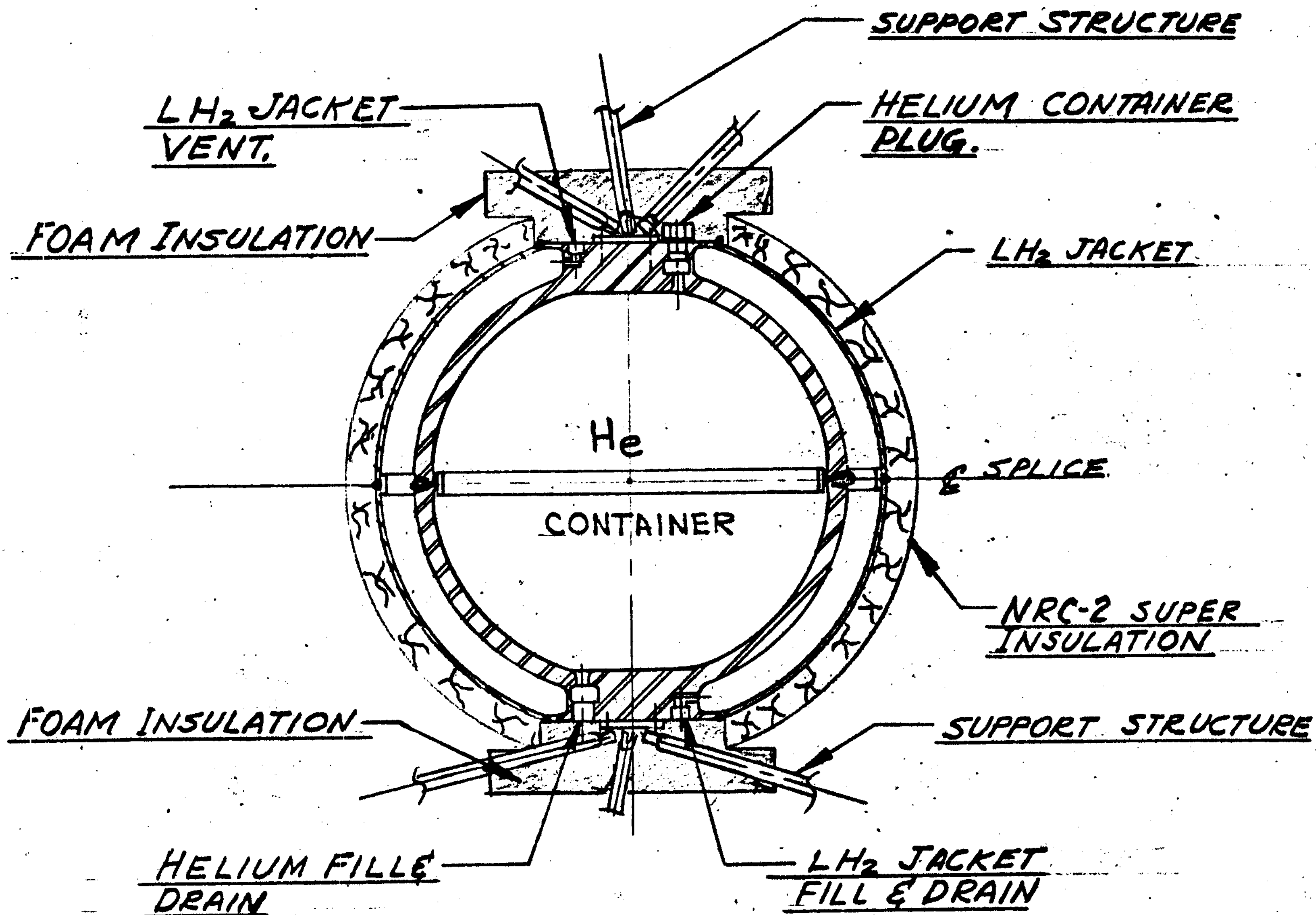
FIGURE 74 CRYOGENIC JACKETED He STORAGE CONTAIN.



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circulation of liquid hydrogen through the jacket is required.

If air is allowed to circulate between the layers of NRC-2 superinsulation, condensation, and eventually solidification of air will result. It is therefore necessary to exclude air from the superinsulation prior to launch while the vessel is in a cooled condition. The proposed method of accomplishing this is by purging the insulation with helium. A lightweight, flexible bag would surround the insulation to receive helium from a ground supply. The supply flow rate will be adjusted to overcome the effects of leakage from the bag, to the extent that a slight pressure above ambient is maintained in the bag. An alternate method of solving the air liquification problem is to provide an evacuated metal jacket around the insulation. The addition of this jacket would increase the system mass by about 40 lb_m.

Immediately prior to launch, the remaining liquid hydrogen in the jacket is drained. Also, the helium purge is disconnected. Then, during the launch phase, the remaining helium within the insulation will bleed out normally until a vacuum condition is attained.

Pertinent design data are given in Table 30.

Table 30

Primary Storage Container			
Sphere Volume	=	6.8	ft^3
Inside Diameter	=	28.3	inches
Sphere Weight	=	66.1	lb_m
Helium Loaded Weight	=	71.7	lb_m
Helium Expelled Weight	=	68.3	lb_m
Helium Storage Temp.	=	37	°R
Nominal Loading Pressure	=	2000	psia
Design Maximum Pressure	=	2200	psia (at 530°R)
Minimum Pressure	=	400 psia	
Insulation Weight	=	8.0	lb_m
Insulation Thickness	=	2.0	inches
LH_2 Jacket Capacity	=	30.	lb_m
LH_2 Jacket Weight	=	15.	lb_m
LH_2 Jacket I.D.	=	35.8	inches
Cascade Storage Container			
Sphere Volume	=	2.75	ft^3
Inside Diameter	=	21.0	inches
Sphere Weights	=	53.4	lb_m
Helium Loaded Weight	=	6.7	lb_m
Helium Expelled Weight	=	5.5	lb_m
Helium Storage Temperature	=	530°R	
Nominal Loading Pressure	=	4000	psia
Minimum Pressure	=	450	psia
Design Maximum Pressure	=	4400	psia

B. HEAT TRANSFER ANALYSIS

The design of the primary pressurant storage system is constrained by two thermal considerations:

1) helium mass requirements - hence storage container size - increases with decreasing heating rates, and
2) limit operating pressure - hence wall thickness and sphere mass - increases with increasing heating rates.

These constraints necessitate that the pressurant storage system be sized for the minimum anticipated heating rate, but must be stressed to sustain the maximum incurred heating rate. The thermal analysis of this system, summarized briefly below illustrates the approach used to meet these constraints.

The heat transfer analysis conducted for the selected system can be divided into three parts. The three parts are the heat transfer analysis on the complete pressurant storage system, the heat transfer analysis for designing the pressurant storage system, and the heat transfer analysis for obtaining the maximum heat leak for the designed system.

The heat transfer analysis for the complete pressurant storage subsystem considered heat transfer through the tank supports, insulation and all tubing and lines connected to the sphere. The super insulation thermal conductivity (4.932×10^{-8} Btu/sec ft^2 °R) used for this analysis was considered applicable for the flight configuration. This analysis predicted a heat transfer rate of approximately 13.0 Btu per hour.

For the design of the pressurant storage subsystem, i.e., the helium storage container weight and helium loaded weight for both primary and cascade systems, a different approach was considered for the heat transfer analysis. The approach was to use the minimum heat transfer rate that could be obtained to design the pressurant storage subsystem. This approach would produce a conservative size for the pressurant storage subsystem. This heat transfer analysis considered heat transfer through the tank supports and insulation. A minimum value for the superinsulation thermal conductivity (0.4932×10^{-8} Btu/sec ft^2 °R) was used for this analysis. The analysis predicted a heat transfer rate of approximately 3.0 Btu per hour. The pressurant storage subsystem was, therefore, designed for the minimum heat transfer rate of 3.0 Btu per hour.

After a sphere size and helium loaded weight was obtained for pressurant storage subsystem, a parametric study was performed to determine the maximum heat transfer rate that the designed pressurant storage subsystem could withstand without exceeding the limit operating pressure of the primary storage container for the design mission. The results of the study showed that the pressurant storage container could withstand a 30 Btu per hour heat leak without exceeding the limit operating pressure. The 30 Btu per hour heat leak was an average value over the entire design mission time.

The heat transfer analysis conducted for the detailed analysis portion of Phase I can be summarized as follows:

1) A minimum heat transfer rate of 3 Btu per hour was obtained and used to size the pressurant storage subsystem to obtain a conservative sphere size.

2) The predicted heat leak through the tank supports, insulation, and tubing for the pressurant storage subsystem is approximately 13. Btu per hour.

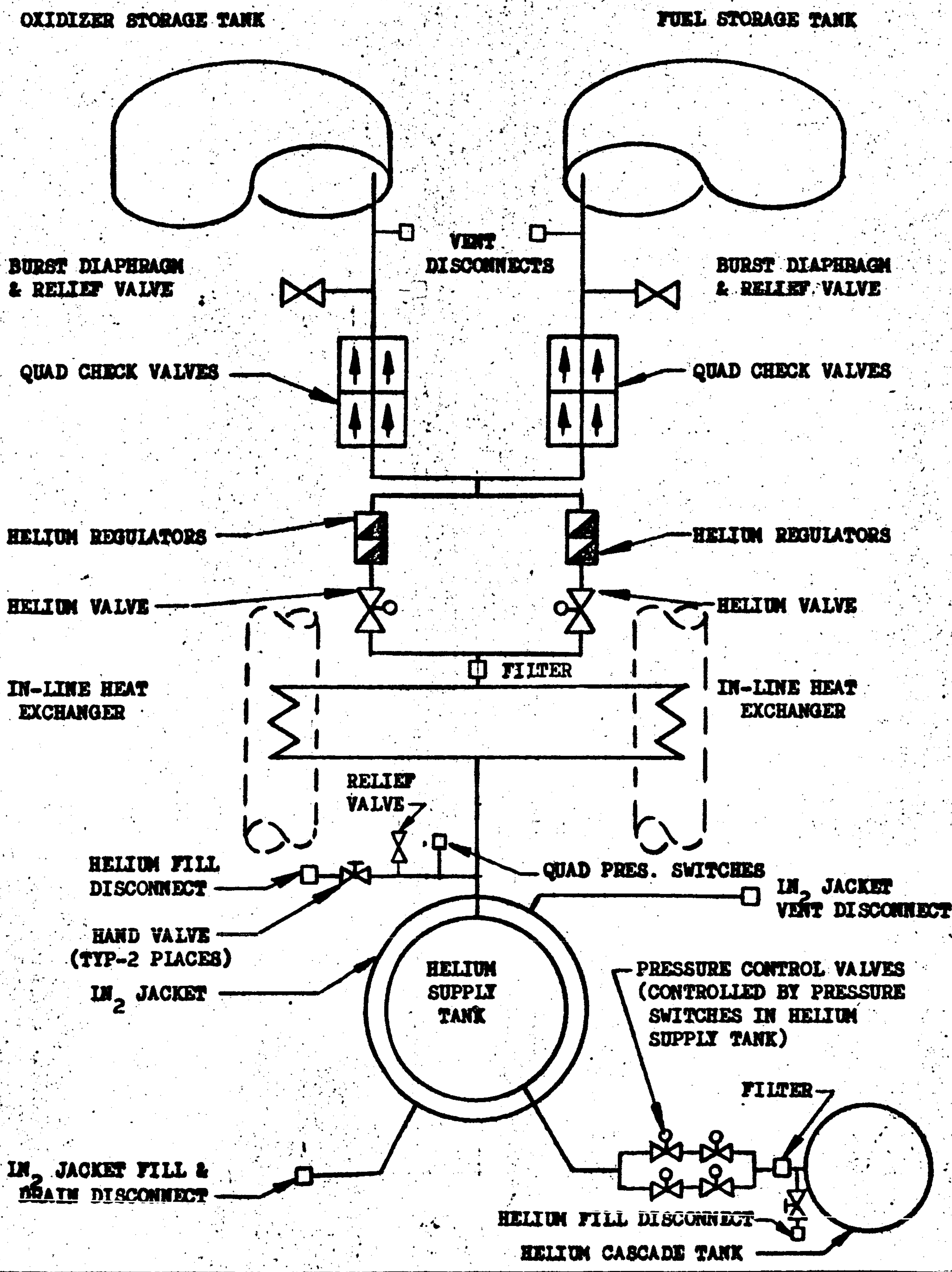
3) The pressurant storage subsystem can operate successfully for heat transfer rates of 3 to 50 Btu per hour.

C. USE OF EXISTING APOLLO COMPONENTS

A brief study was made to determine the impact of using the existing Apollo regulation components in conjunction with the selected system. This might prove advantageous in regard to the cost and development schedule required for implementing a new system.

It was immediately apparent that the helium must pass through the propellant feed line heat exchangers prior to entering the regulators. This is necessary because the regulators are not designed to operate properly with helium at liquid hydrogen temperature. This requires the use of a high pressure (2000 psia) heat exchanger, rather than the low pressure unit now used. A schematic of this system (designated as system 5A) is shown in Figure 75. The system upstream of the heat exchangers is identical to the proposed system 5, described previously. Downstream of the heat exchangers, system 5A is identical to the existing Apollo pressurization system.

The additional analyses performed for this system were resizing of the heat exchangers, evaluating total system mass, and determination of reliability. Previous analyses for the helium storage system and helium usage requirements were entirely valid



ADVANCED LIGHTWEIGHT PRESSURIZATION SYSTEM CASCADE HELIUM STORAGE SYSTEM PASSIVE FLOW HEATING	SYSTEM NUMBER 5A

DRAWN BY: A.F. Bingham 7-65 APPROVED: D.H. Horman 7-65	THE MARTIN COMPANY DENVER	FIG 75 CHG A PAGE

for this system concept. A comparison of mass and reliability for systems 5 and 5A are shown below.

	System 5	System 5A
Mass	344 lb_m	383 lb_m
Reliability	.999780	.999688

Additional evaluation will be required in order to compare the increase in weight and loss in reliability with the apparent reduction in system development cost which can be achieved by adopting Apollo regulation.

VII. CONCLUSIONS

1. Hydrogen as a Pressurant for the Apollo System

For the Apollo system, mass of usable pressurant can be reduced only by minimization of pressurant molecular weight. Gas molecular weight can be decreased only by replacing the existing pressurant, helium, with hydrogen; thus reducing molecular weight from 4 to 2. This change was analyzed for the fuel tank and it was found that the system mass thus saved was almost equally offset by resulting increases in pressurant storage container mass.

2. Reduction of Residual Gas Weight

Three methods of increasing residual gas temperature (thereby reducing residual mass) were considered:

a) use of an internal heat exchanger, using a liquid propellant hot gas generator heat source,

b) use of solid propellant sodium azide gas generator to expel hot nitrogen directly into the pressurant storage tank, and

c) introducing warmer helium directly into the storage tank.

The latter method is known as the cascade concept, and depends upon the cascade helium supply being stored at a significantly higher temperature than the primary tank. The cascade system concept is by far the lightest of the three mentioned. Concept a) has a potential of being lighter if a "free" source of heat could be obtained from a different vehicle subsystem, but the use of a special gas generator and propellant supply more than offsets the weight savings of reducing residuals.

3. Gas Generator Systems

Systems utilizing hydrazine monopropellant gas generator gases as fuel tank pressurants are very efficient from a weight standpoint. A system of this type could be designed for the Apollo service module for 280 pounds.

4. Cryogenically Stored Helium

A weight reduction of 370 pounds can be achieved by reducing the gas storage temperature of an Apollo-like system to 37°R. However, most of this weight reduction, 341 pounds, can be achieved by reducing storage temperature to 140°R

5. The Recommended System

The three best systems studied were the gas generator system (8), the cryogenically stored helium system (1), and the cascade system (5). The cascade system produces the greatest weight savings and is satisfactory from every other aspect considered; it is, therefore, the recommended system. (The final comparison of the systems is shown in Table 29).

6. The Possible Weight Savings

The incorporation of system 5, the cascade system, into the Apollo Service Module would result in a 535 pound savings. An additional seventy pound savings could be achieved at the same time by re-sizing the storage system for the actual gas weight now being planned.

APPENDIX A

Reports issued in conjunction with Advanced Lightweight Pressurization System Contract No. NAS 9-3521, excluding monthly progress reports.

Martin CR-64-78	Planning Report	October, 1964
Martin CR-64-80	Survey of System Concepts	November, 1964
Martin CR-64085	Propellant Tank Internal Mass and Heat Transfer Analysis Techniques	December, 1964
Martin CR-64-78 (Issue 2)	Planning Report	January, 1965
Martin CR-65-3	General Test Plan	December, 1964
Martin CR-65-6	System Selection Summary for Advanced Lightweight Pressurization System	January, 1965
Martin CR-65-9	Preliminary Mass Comparison Study for a Helium-Sodium Azide Pressurization System for the Apollo Service Propulsion System	February, 1965
Martin CR-65-10	Utilization Instructions - Tank Pressurization Computer Program ØDO41	February, 1965
Martin CR-65-12	Criteria for Advanced Lightweight Pressurization System Prototype Selection and Optimization	February, 1965
Martin CR-65-3 (Issue 2)	Experimental Test Plan	March, 1965
Martin CR-64-78 (Issue 3)	Planning Report	April, 1965
Martin CR-65-3 (Issue 2A)	Addendum to Experimental Test Plan	March, 1965
Martin CR-65-36	Preliminary Utilization Instructions Cascade Pressurization Computer Program	May, 1965
Martin CR-65-37	Preliminary Utilization Instructions Gas Expansion Computer Program	May, 1965

APPENDIX B

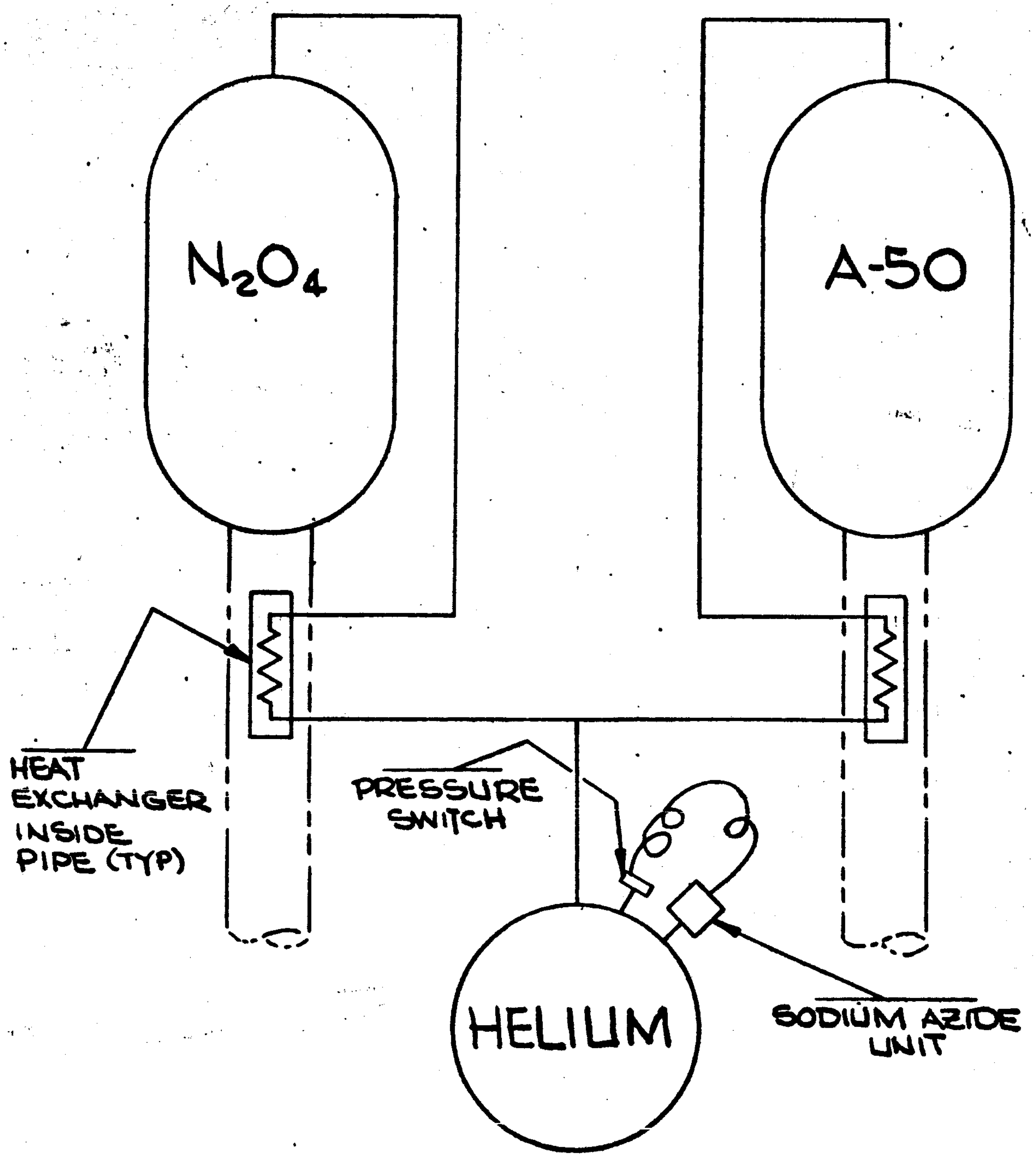
Two additional concepts of propellant tank pressurization were considered as possible candidates during the early efforts in the detailed design and analysis part of Phase I. However, preliminary analytical investigation of those concepts indicated that none was potentially capable of saving significant weight in the present Apollo Service Propulsion System. The two concepts are discussed below.

1. Helium - Sodium Azide Concept

The use of a sodium azide (NaN_3) gas generant was investigated as a heat source within the storage tank of an ambient stored helium pressurization system. The objective of this study was to determine if the mass of the ambient stored helium system could be significantly reduced by such a modification.

The helium-sodium azide system is shown schematically in Figure 1B. It is identical to the standard ambient stored helium system, with the addition of one or more sodium azide gas generator units. Helium is expanded from the storage container in the normal manner until an established minimum pressure is reached (400 lb_f/in^2 absolute for the Apollo SPS system). At this point an azide unit is ignited, adding heat and mass to the stored helium thereby increasing the pressure to a desired upper level. The gas mixture now in the helium storage container expands into the propellant tanks until the minimum pressure is again reached, another azide unit is ignited, and the process is repeated until the propellant tanks are emptied.

The combustion products of the sodium azide gas generator are composed of about 56.5% gas and 43.5% solid particles, by mass measurement. The gas consists of 99.5% nitrogen and .5% hydrogen,



TITLE: HELIUM - SODIUM AZIDE PRESSURIZATION SYSTEM	NUMBER FIGURE 1B
THE MARTIN COMPANY DENVER	PAGE	CHG.

also by mass.[1] The main constituents of the solid matter generated are sodium, carbon, and sodium fluoride. Since hydrogen is such a small percentage of the gas generated, the gas is considered to be composed only of nitrogen for purposes of this preliminary study.

For purposes of computing the azide system mass, it is considered that the mass of gas (N_2) generated represents 50% of the initial mass of the gas generator unit. The remaining 50% is inert mass, consisting of the solid matter generated and structural mass of the gas generator unit.[1]

Included herein is a mass comparison of a normal ambient stored helium system (as used in the current Apollo SPS) and a helium-sodium azide system. The comparison is on the basis of pressurant plus pressurant storage system only. Other components, such as tubing, valves, etc., should weigh about the same for both systems.

Mass of the reference ambient stored helium system was taken from previous preliminary study data as 654 lb_m.[2] Mass of the helium - azide system was parametized about two variables; mass of helium loaded, and storage container final temperature. Mass of helium load was varied from the amount required for complete propellant tank pressurization, down to about half that amount. As helium mass decreased, it was necessary to use more nitrogen for propellant tank pressurization. Final temperature fo the storage container was varied from ambient (530°R) to 1000°R. Actually it is not considered feasible to allow the gas storage temperature to rise above

1. Letter from Aerojet-General Corporation (W. J. Flaherty), to D. N. Gorman. 21 December 1964.

2. "System Selection Summary for Advanced Lightweight Pressurization System." Table 1, page III-15. Martin CR-65-6. Contract NAS 9-3521. January, 1965.

600°R, due to maximum operating temperature limitations of the quantity gaging system within the propellant tanks. However, the higher storage temperatures were included in this study for comparative purposes.

It was assumed that all helium loaded was utilized as propellant tank pressurant; residual gas in the storage container was therefore entirely nitrogen, existing at the minimum pressure of 400 lb_f/in^2 absolute and an arbitrary temperature. Since the independent variables in this analysis were selected as mass of helium used and storage container final temperature, the mass of nitrogen required and the azide system inert mass must be calculated. The nitrogen mass was determined as the sum of the following:

M_{N_2} = mass of fuel tank nitrogen pressurant + mass of oxidizer tank nitrogen pressurant + mass of residual nitrogen in pressurant storage container.

or,

$$M_{N_2} = \left(\frac{P_{N_2} V}{R_{N_2} T}\right)_{\text{fuel tank}} + \left(\frac{P_{N_2} V}{R_{N_2} T}\right)_{\text{oxid. tank}} + \left(\frac{PV}{R_{N_2} T}\right)_{\text{storage container}}$$

where, P_{N_2} = partial pressure of nitrogen at end of mission,

V = volume of gas at end of mission,

R_{N_2} = specific gas constant, nitrogen,

T = temperature of gas at end of mission

The partial pressure of nitrogen in each tank was calculated by

$$\left(P_{N_2}\right)_{\text{fuel tank}} = P_T - P_{He}$$

$$= 175 - \left(\frac{M_{He} R_{He} T}{V}\right)_{\text{fuel tank}}$$

and $P_{N_2 \text{ oxid. tank}} = P_T - P_{He}$

$$175 - \left. \frac{M_{He} R_{He} T}{V} \right|_{\text{oxidizer tank}}$$

where, P_T = propellant tank pressure (175 lb_f/in^2 absolute),

P_{He} = partial pressure of helium at end of mission,

M_{He} = mass of helium in propellant tank at end of mission,

R_{He} = specific gas constant, helium,

T = temperature of gas at end of mission,

V = volume of gas at end of mission.

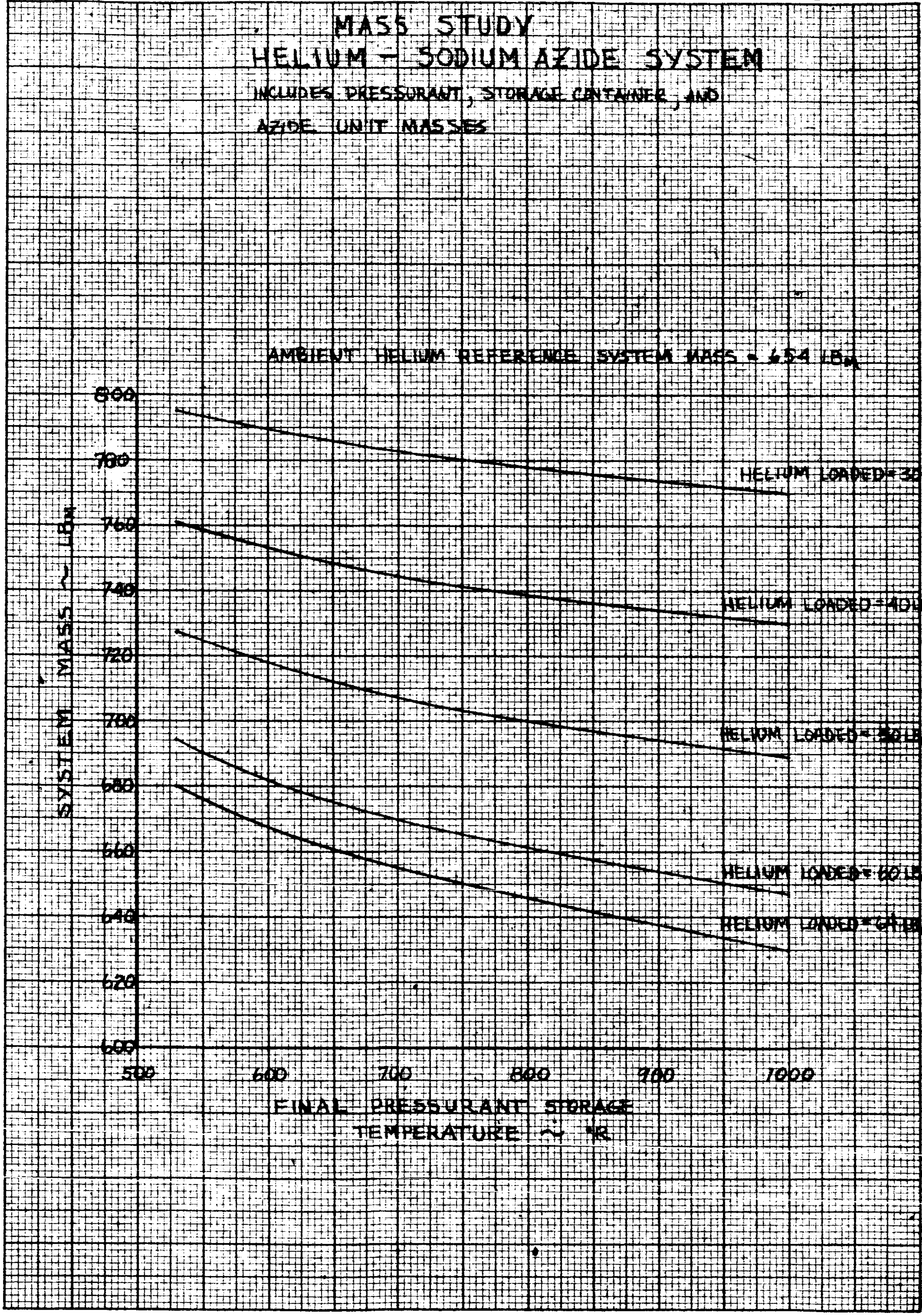
As stated previously, the sodium azide inert mass is taken as being equal to the mass of nitrogen generated. The total system mass is then defined as

System mass = helium mass + storage container mass + nitrogen mass + sodium azide inert mass.

The results of this study are plotted in Figure 2B. System mass increases as the mass of helium loaded decreases; this is because the mass of nitrogen plus azide system inerts more than offsets the savings in helium and storage container mass. As Figure 2B illustrates, the optimum utilization of a helium-sodium azide pressurization system requires designing the storage container to hold sufficient helium for complete propellant tank pressurization; the azide unit is then used only to provide minimum storage container operating pressure.

It is noted in Figure 2B that the helium-azide system mass is greater than an ambient stored helium system at final storage temperature below 700°R. At temperatures above 700°R the helium-azide

Figure B-2


MASS STUDY
HELIUM - SODIUM AZIDE SYSTEM
INCLUDES PRESSURANT, STORAGE CONTAINER, AND
AZIDE UNIT MASSES
AMBIENT HELIUM REFERENCE SYSTEM MASS = 654 LBm
SYSTEM MASS ~ LBm
800
780
760
740
720
700
680
660
640
620
600
HELIUM LOADED = 30 LBm
HELIUM LOADED = 40 LBm
HELIUM LOADED = 50 LBm
HELIUM LOADED = 60 LBm
HELIUM LOADED = 64 LBm
500
600
700
800
900
1000
FINAL PRESSURANT STORAGE
TEMPERATURE ~ °R

system becomes lighter than the ambient helium system, but at a very gradual rate. At 1000°R, the helium-azide system is only 21 lb_m lighter than the ambient helium system.

It has been established that the maximum temperature allowable in the propellant tank ullages is 600°R, due to design specification limitations on the propellant gaging system. Mass of the helium-azide system at a storage temperature of 600°R is 668 lb_m, or 14 lb_m above the ambient stored helium system.

It is concluded that an optimized helium sodium azide pressurization system for the Apollo Service Propulsion System would weigh about the same as the present ambient stored helium system. Furthermore, the addition of the required azide units, and pressure sensing device would tend to reduce system reliability significantly. Therefore, gas generation by sodium azide units will receive no further consideration in this program.

Mass estimates of the helium-sodium azide system were based upon information supplied by Aerojet-General Corporation in December, 1964. Further information on the sodium azide units is included in "Bulletin of the Second Meeting of the Joint Army-Navy-Air Force Liquid Propellant Group." pp 645-682. November, 1960. Information on an ammonium nitrate gas generant was also received from Aerojet-General Corporation. The ammonium nitrate unit has a higher mass fraction (about .85) than the azide, but produces excessive amounts of hydrogen gas in the combustion products. Hydrogen constitutes about 30%, by volume, of the combustion products. This high concentration of hydrogen would form an explosive mixture with the N_2O_4 vapor in the oxidizer tank; therefore, the use of ammonium nitrate was considered too hazardous for application to the Apollo SPS pressurization system.

2. Vaporized Liquid Pressurization Concept

An analytical inquiry was made into the feasibility of using some form of "vaporized liquid" pressurization system. This concept was felt to be potentially applicable to either tank as a modification to present system No. 1 or to the oxidizer tank as a modification to present system No. 8.

Figure 3B shows estimated pressurant usage requirements for both fuel and oxidizer tanks as a function of pressurant molecular weight. Final propellant tank temperatures were assumed to be a nominal 70°F. The reference mass of an ambient stored helium system for the Apollo Service Propulsion System (SPS) is 654 lb_m for pressurant and storage container. Of this, approximately 355 lb_m can be charged to oxidizer tank pressurization, and 299 lb_m to fuel tank pressurization. If storage system mass for a vaporized liquid system is taken as 10% of pressurant mass, and a 5% loading margin is added as residuals, it is determined that the vaporized liquid system will be lighter than an ambient stored helium system only if oxidizer pressurant mass is below 308 lb_m and fuel tank pressurant mass is below 259 lb_m. This corresponds to pressurant molecular weights of 34 for the fuel tank and 35 for the oxidizer tank (Figure 3B).

Considering first the fuel tank of system 1, efforts were made to identify possible pressurants. Only compounds which were deemed as "fuel-like" (i.e., unlikely to undergo chemical reaction with the fuel in the propellant tank) were considered. For the fuel tank, ammonia (molecular weight of 17) appeared to be the only

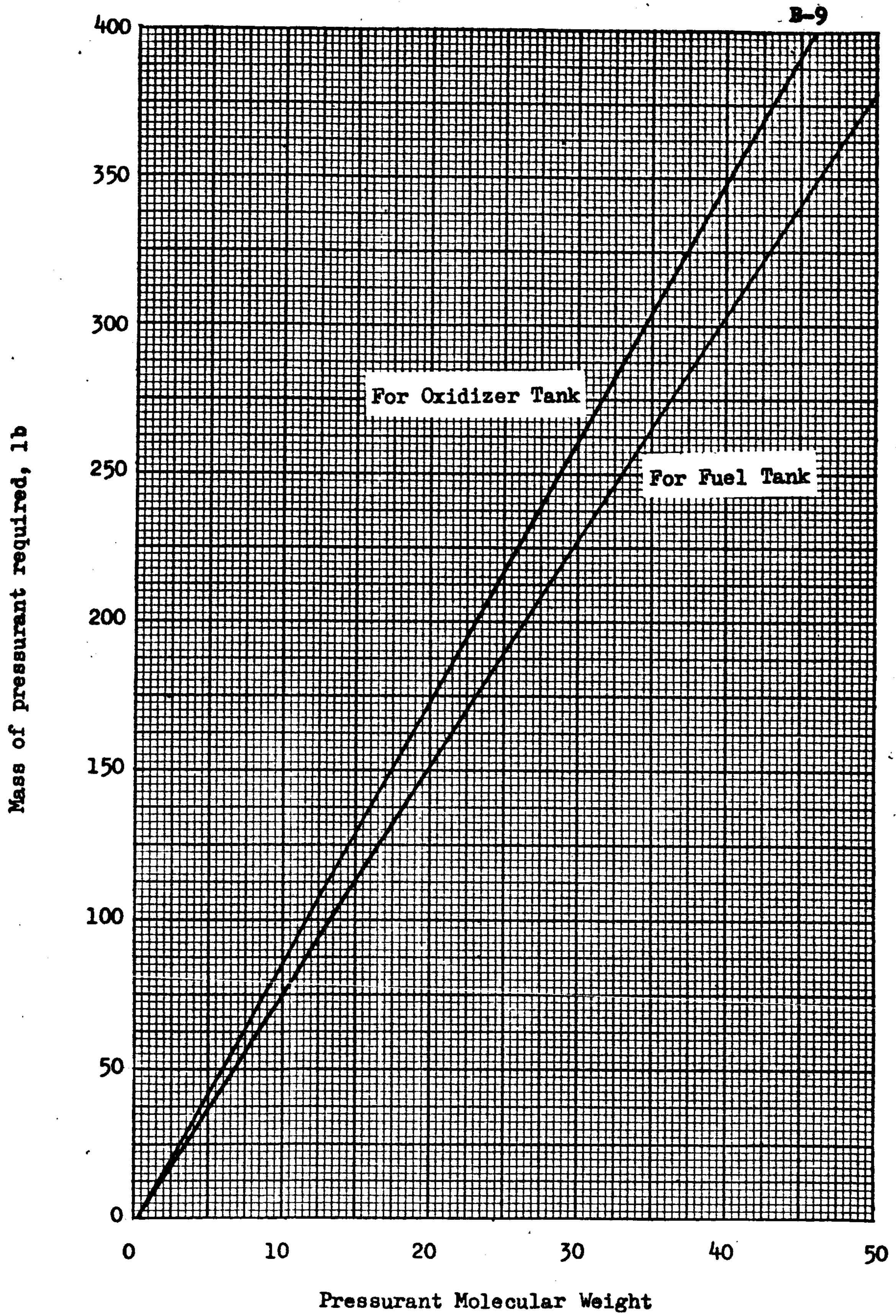


Figure 3B- MASS OF PRESSURANT REQUIRED TO PRESSURIZE APOLLO SPS PROPELLANT TANKAGE (PRESSURANT CONDENSATION NOT CONSIDERED) compound potentially suitable, due to its comparatively low molecular weight of 17. Further examination, however, indicated that the unusually steep rate of change of vapor pressure over the limiting propellant temperatures of 40 to 80°F could cause significant uncertainty in mass of condensed pressurant. Certian characteristics of an ammonia-pressurant system are displayed in Figure 4B.

For the oxidizer tank, no suitable pressurants could be identified.

A modified version of system 8 was also examined during this reporting period. The original version of system 8 (which remains as a candidate system at the present time) employs stored helium as the ozidizer tank pressurant -- this helium is heated in an active heat exchanger prior to delivery to the oxidizer tank. Combustion products from an Aerozine-50/N_2O_4 gas generator serve as the hot fluid for the heat exchanger and this hot gas exiting from the heat exchanger is used as the fuel tank pressurant. (Note: the original weight analysis for System 8 was discussed on page II-38 through II-42 of the Monthly Progress Report for January 1965).

The modified version of system 8 is similar to the original version except that a mixture of nitric oxide (NO) and nitrogen tetroxide (N_2O_4) is used both as the oxidizer tank pressurant and also as the gas generator oxidizer. Mixtures ranging in composition from 30% NO - 70% N_2O_4 (by weight) to 60% NO - 40% N_2O_4 were selected for examination for vapor pressure and molecular weight considerations. The NO-N_2O_4 mixture was considered to be stored as a liquid in a storage sphere -- pressurization of this storage

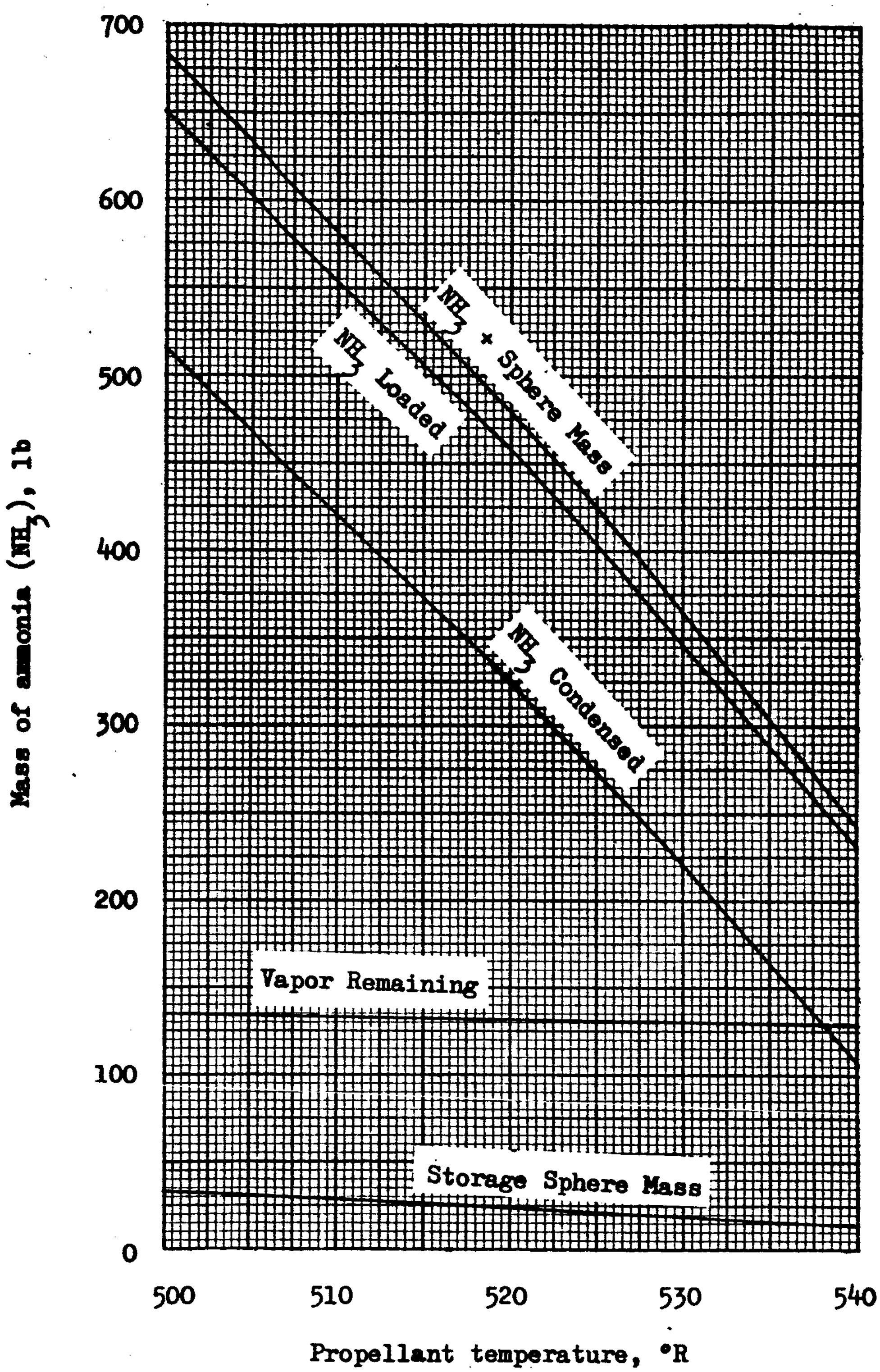


Figure 4B- MASS OF AMMONIA REQUIRED FOR FUEL TANK PRESSURIZATION sphere was by separate stored helium system. The mixture was delivered to the heat exchanger wherein it was heated into the gas-phase, thence delivered to the oxidizer tank.

The analysis technique employed for the modified system 8 followed that technique developed for the previous analysis effort. Pressurant usage for the oxidizer tank was established by use of the system mathematical model, ØDO41. Sizing of (and weight of) the $NO-N_2O_4$ storage sphere was established by hand calculation.

The pressurant for the fuel tank is the hot gas exhaust from the heat exchanger and the temperature of this hot gas is dependent upon the amount of energy delivered to the $NO-N_2O_4$ in the heat exchanger.

The hot gas mass flow rate requirements are thus again seen to be dictated by two considerations:

1) the energy needed to vaporize the $NO-N_2O_4$ mixture in the heat exchanger, and
2) the mass flow and temperature requirements at the fuel tank top.

Mass flow rate requirements for the $NO-N_2O_4$ mixtures used as oxidizer tank pressurant were determined by use of the ØDO41 program. Consideration was given to the condensation of N_2O_4 from the pressurant mixture. Corresponding required hot gas flow rates through the heat exchanger (i.e., evaporator) were then calculated. It was seen (again, as the previous system 8 study) that the hot gas flow, as dictated by fuel tank pressurant needs, exceeded the hot gas flow required by the heating (and phase change) of the $NO-N_2O_4$ mixture.

Calculated system weights for this modified version of system 8 are shown in Table 1B. Apollo system reference mass with which Table 1B should be compared is 698 lb_m, as established in the preliminary analysis (Section III of this report).

Table 1B - System Weights for Modified System 8

Component	Component weights... ...for oxidizer tank pressurant composed of 30% NO/70% N_2O_4	for oxidizer tank pressurant composed of 60% NO/40% N_2O_4
N_2O_4 required	229.6	251.7
NO required	535.8	167.8
Gas generator propellant required	178	92
Gas generator weight	8	8
Valves	39	39
Heat exchanger	16	16
Storage vessel for GG propellants	11	6
Storage vessel for NO/N_2O_4 mixture	27	20
Pressurization system for NO/N_2O_4 storage vessel	9	7
Total System Weight	1053	607

This modified version of system 8 was examined by the NASA Technical Representative and the Martin Company and, by concurrent agreement, was removed from any further consideration since no significant advantage for the system was indicated.